FINAL TECHNICAL REPORT

GEODETIC SPACECRAFT

(28 February 1961 to 30 October 196)

FTR/68-1

Contract DA-44-009-ENG-4759

Department of the Army, Project No. 8T35-11-0 -10

Corps of Engineers

Placed by

U. S. Army Engineer Geodesy, Intelligence
and Mapping Research and Development Agency
Fort Belvoir, Virginia

CUBIC CORPORATION
5 5 Kearny Villa Road
San Diego 11, California

Page	Paragraph	Change
2-2	2.3.2	Change item (1) to read: IM/59-1 Instruction Manual for Geodetic SECOR, Volume I, System Description, 1 January 1961.
		Change item (2) to read: IM/59-1 Instruction Manual for Geodetic SECOR, Volume II, RF Equipment, 1 June 1961.
3-4	3.4.3.1	Change 8th sentence to read: The separation fixture used on the flight models is surface polished to less than 4 micro inches, and coated with silicon monoxide for thermal stabilization.
3-10	3.4.5.1	Change last sentence to: These were modified, and the splice channel was tapped in order to facilitate assembly and disassembly.
3-12	3.5.3	Last sentence: Change "570 mc" and "575 mc" to "570 kc" and "575 kc", respectively.
3-16	3.5.4	(1), (2), (3), (4): Change "570 mc" and "575 mc" to "570 kc" and "575 kc", in order shown.
3-28	3.5.5.3	Change step (21) to read: The HF frequency shall be 449.000 mc ±0.0025 per cent.
3-32	Figure 3-15	Change: "Z_o = 3 ±1V" to "E_o = 3 ±1V"
3-33	3.6.3.1.2	Change 5th sentence to read: The non-separation signal is provided by a commutator clock pulse which is connected to channel No. 3 (battery voltage) and powered through the microswitch.

Page	Paragraph	Change
3-63	Figure 3-34	Change figure number to 3-35, and change "Before Adjustment" to "After Adjustment".
3-64	Figure 3-35	Change figure number to 3-34, and change "After Adjustment" to "Before Adjustment".
3-67	3.7.2.3	Change the first sentence to read: To provide a 12V system with nickel-cadmium cells, it was necessary to series-connect ten such cells, to form one "bank". Change the second sentence to read: To insure reliable operation, two banks are connected in parallel; each bank can handle the entire intermittent load.
3-85	3.8.3.1	Change first sentence to read: The 136 mc antenna system is a one-quarter wavelength monopole located on the north pole of the satellite.
3-90	3.8.4.1	Change fifth sentence to read: The 136 mc antenna system is a single 1/4-wavelength monopole attached at the north pole of the Spacecraft.
3-144	3.8.5.2.2	Change fourth sentence to read: The antenna immediately adjacent to the flight plug has been modified to fold at an angle 16.5 degrees above the equator of the Spacecraft.
3-157	3.10.1	Last line. Change to read: thermal gradient across the Spacecraft.
3-161	3.11.1.2	In the 4th sentence, change: $\alpha / \varepsilon < 1$ to $\alpha / \varepsilon \ll 1$.

Page	Paragraph	Change
3-169	3.11.3.1.10	Second line. Change to read: discussion it appears that a better design approach would be to allow the
3-171	3.11.3.2.3	Definition of terms, equation (3.16), 2nd expression: Change $\Delta T(x)$ to $\Delta T(0)$.
3-186	3.13.3	Change step (2) to read: Test items and test mount equalized to within ±1.5 db and subject to the following random vibration for a period of four minutes.
3-189	3.13.3	Top line on page 3-189. Delete: "for four minutes".
3-197	3.13.4.1.1	Add the following at the end of last paragraph: Since a resonant condition was noted in the frequency range of 30-32 cps, it was decided that the separation fixtures would be modified on serial numbers 2 and 3. Refer to paragraph 3.4.3.1 for modification details.
3-277	Table XXII	First column, fourth box. Change: NORMAL to LOAD.
		Second column, fourth box. Change the function opposite SHORT to read: When the switch is in SHORT position, solar cells are shorted to ground in order to measure short circuit current.
		Second column, fourth box. Change the function opposite LOAD to read: When switch is in LOAD position, solar cell open-circuit voltage or charging current may be measured.
4-1	4.2.1	Fifteenth line of paragraph. Change: "174 cw in." to "174 cu in."

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CUBIC CORPORATION, 5575 Kearny VillaRoad, San Diego 11, California

Report FTR/68-1, 30 October 1961, 341 pp, 117 illus, 27 tables
DA Project 8T35-11-001-10 Unclassified Report

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1. Spacecraft Development, Geodesy, Mapping and Charting
2. Contract - DA-44-009 eng- 4759

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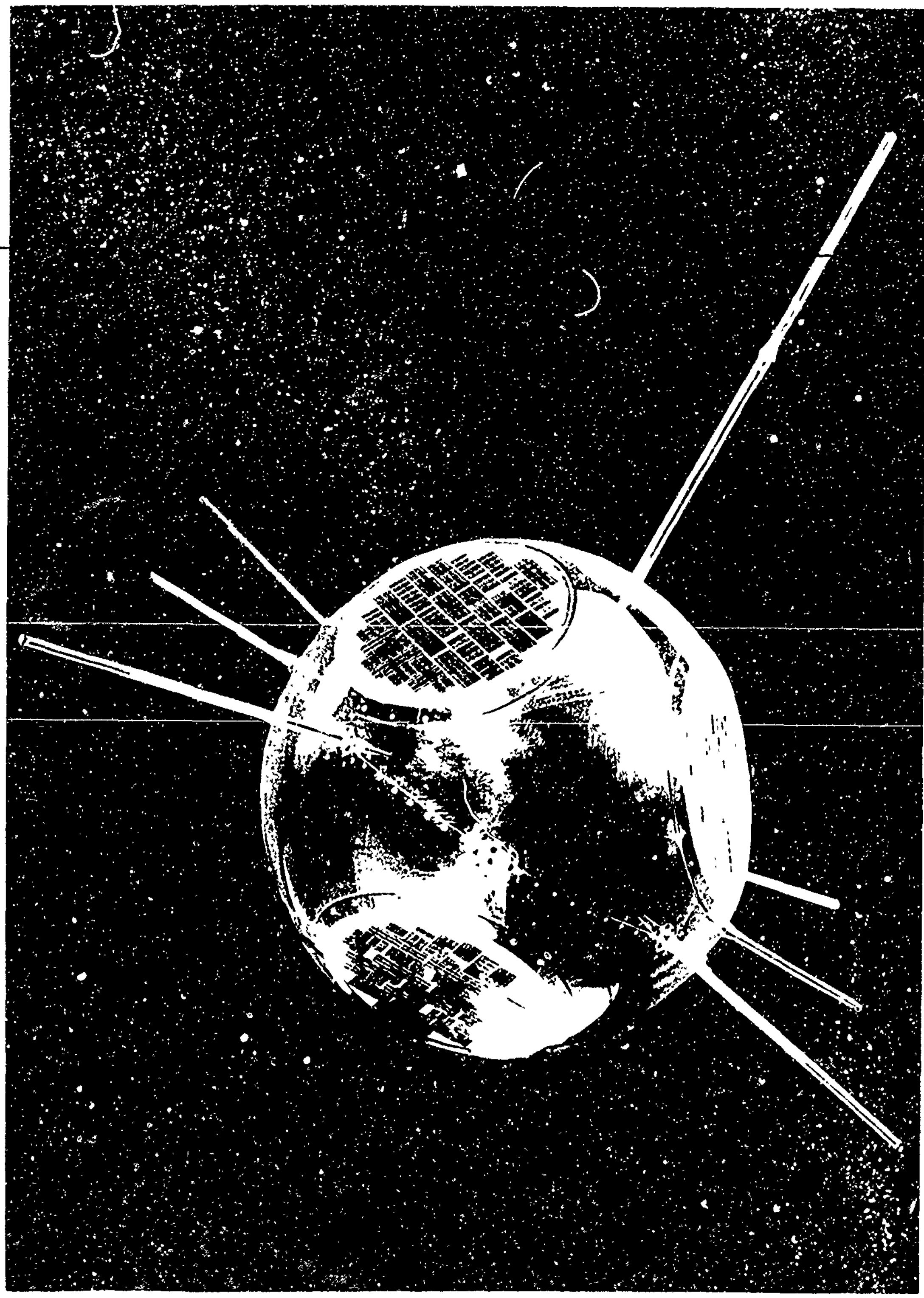

Frontispiece. Cubic Geodetic Spacecraft

PREFACE

Department of the Army Contract DA-44-009-ENG-4759 covers the design, fabrication, assembly, test, and preparation for launch of three Geodetic Spacecraft for the U. S. Army Engineer Geodesy, Intelligence and Mapping Research and Development Agency (GIMRADA), Fort Belvoir, Virginia, by Cubic Corporation San Diego, California. This contract was accomplished during the period from 28 February 1961 to 30 October 1961.

Monthly progress reports were submitted throughout the contractual work period. These reports outlined the preliminary study and design investigations, and described the relevant details of fabrication, assembly, and testing including discussions of the technical problems encountered. It is required that the complete history of all the work accomplished be summarized and documented in this Final Technical Report.

The Geodetic Spacecraft configuration as presented in this report is designed to operate in conjunction with the Distance Measuring Equipment (DME) of the Geodetic SECOR System manufactured by Cubic Corporation. The integrated system will be utilized to conduct geodetic measurements. The spacecraft carries and powers a Cubic SECOR TR-17 or TR-27 transponder that receives composite data frequencies on a 420. 9375 mc DME carrier and retransmits the data frequencies at 224. 5 mc and at 449. 0 mc. In addition, a solid-state telemeter reports nine "housekeeping"-type measurements on IRIG band No. 3 at 136. 11 mc, and also serves as a beacon transmitter for initial acquisition and computation of orbital parameters. The spacecraft contains batteries and solar cells sufficiently redundant to ensure adequate power during the proposed year in orbit. The Geodetic Spacecraft will be injected into orbit on the Composite I Multiple Satellite launching aboard a Thor Able Star vehicle.

ACKNOWLEDGEMENTS

Cubic Corporation wishes to acknowledge the assistance received from the following organizations during the conduct of this program. Their cooperation is greatly appreciated and has been of material assistance in the successful completion of this phase of the program.

1. Goddard Space Flight Center of the National Aeronautics and Space Administration (NASA), Greenbelt, Maryland

2. Naval Research Laboratories (NRL), Washington, D. C.

3. U. S Army Engineer Research and Development Laboratories (ERDL), Fort Belvoir, Virginia

4. Applied Physics Laboratory (APL), Johns Hopkins University, Silver Springs, Maryland

5. Brooks and Perkins Corporation, Detroit, Michigan

6. Hoffman Electronics Corporation, El Monte, California

7. International Rectifier Corporation, El Segundo, California

8. Electronic Balancing Company, Long Beach, California

9. Ryan Electronics Corporation, San Diego, California

10. General Dynamics Corporation, San Diego Division, San Diego, California

REFERENCES

1. Heller, IRE Transactions on Military Electronics, MIL 4, No. 2-3 (1960)

2. Hibbs, "The Temperature of an Orbiting Missile," JPL Progress Report No. 20-294, (1956)

3. Hass, Drummeter and Schach, "Temperature Stabilization of Highly Reflecting Spherical Satellites," Journal of the Optical Society of America, 49, No. 9 (1959)

4. E. P. Buwalda and A. R. Hibbs, "Satellite Temperature Measurements For 1958 Alpha-Explorer I", Jet Propulsion Laboratory External Publication No. 481, 1958

5. Menzel, "Fundamental Formulas of Physics", Prentice Hall, NY, 1955.

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SECTION I
SUMMARY

1.1 Scope of Report. This Final Technical Report presents a complete history of the development and testing of the three Geodetic Spacecraft furnished by Cubic Corporation in accordance with U. S. Army Contract DA-44-009-ENG-4759. The report consists of six sections and three appendices. Section II states the aims and purpose of the contract and describes the fulfillment of contractual requirements. In addition, this section presents a brief description of the related SECOR tracking system with reference to pertinent technical manuals. Section III covers the complete history of the work accomplished during the period of development. The investigation describes the development of the Spacecraft as an integral system and as an assembly of its individual subsystems. Each subsystem is described in terms of its developmental history beginning with the initial design problem and continuing through to validation tests. In addition, section III contains the history of packaging, thermal design, balancing, environmental testing, mass constants, satellite mockup, handling fixtures, special test equipment, and shipping containers. This section presents all pertinent tests and test results, derivation of all related mathematical formulas and a brief description of all difficulties encountered. Section IV describes the operational and technical problems involved in the development of the major subsystems of the Spacecraft. Section V contains the logical conclusions resulting from an analysis of the data derived during the program. Section VI presents the recommendations resulting from the conclusions with reference to the course of proposed future work. The three appendices provide tables of data collected during the thermal-vacuum tests for each Spacecraft.

SECTION II
INTRODUCTION

2.1 Purpose of Contract. The purpose of Army Contract DA-44-009-ENG-4759 is to provide for the design, fabrication, assembly, test, and preparation for launch of three Geodetic Spacecraft. Each Spacecraft shall accommodate a SECOR TR-17 transponder or equivalent, its related power supply and electronics, and shall be utilized in conjunction with the Geodetic SECOR System built by Cubic Corporation, San Diego, California.

2.2 Scope of Report. This report contains a complete history of the design, development, fabrication, and testing of the three Geodetic Spacecraft furnished in accordance with the above contract. The report covers the work accomplished during the period from 28 February 1961 to 30 October 1961.

2.3 Related Programs. The Geodetic SECOR Program provided under Army Contract DA-49-018-ENG-2191 is directly related to the Geodetic Spacecraft Project. A Geodetic SECOR System comprises four distance measuring equipment (DME) ground stations. Each ground station operates on the physical principle that the modulation of a modulated electromagnetic wave propagated through space undergoes a phase shift proportional to the distance traveled and to the modulation frequency. A ground station, at which range is measured, transmits a phase-modulated signal. This signal is received by a satellite-borne transponder which repeats the modulation as phase modulation on an offset carrier frequency. This repeated signal is received at the ground station and the phase shift of the modulation is measured by an electronic servo which provides at its output a digitized representation of range.

2.3.1 Relationship to Spacecraft Geodesy. The over-all concept of distance measuring satellite geodesy is based on a mathematical relationship of sequential distance observations. Two or more DME stations provide information regarding orbital relations

of the satellite to the DME stations. Precise location of each DME station on an accurate continental grid system relates the satellite's orbit to known points on the earth's surface. Using only two DME stations, however, will result in ambiguous data as a result of certain angular combinations of the orbit to the DME stations. Hence, elimination of orbital placement ambiguities is accomplished by using three DME stations located at the vertices of an approximate spherical equilateral triangle. (See figure 2-1.) All three DME stations are referenced to an accurate continental grid system. Any point on the earth's surface which may be referenced to the grid system may be related to the satellite's orbit. The location of an unknown point is established with a fourth DME station. Orbital data of the satellite as determined by the known DME stations is related to orbital observations of the unknown DME station. The location of the unknown DME station may then be referenced exactly to the location of the known DME stations. If line-of-sight conditions exist simultaneously for all four DME stations, simultaneous observations of the satellite range is possible, and location calculations are based on the spatial intersection of distance values, as referenced to a timing system individual to the SECOR system only. However, more economical use of equipment is realized during nonsimultaneous operation; that is, separation of the DME stations beyond line-of-sight conditions. The SECOR system in this mode of operation interpolates orbital data in the "blind" areas, and gains system unity by relating distance measurements to real time.

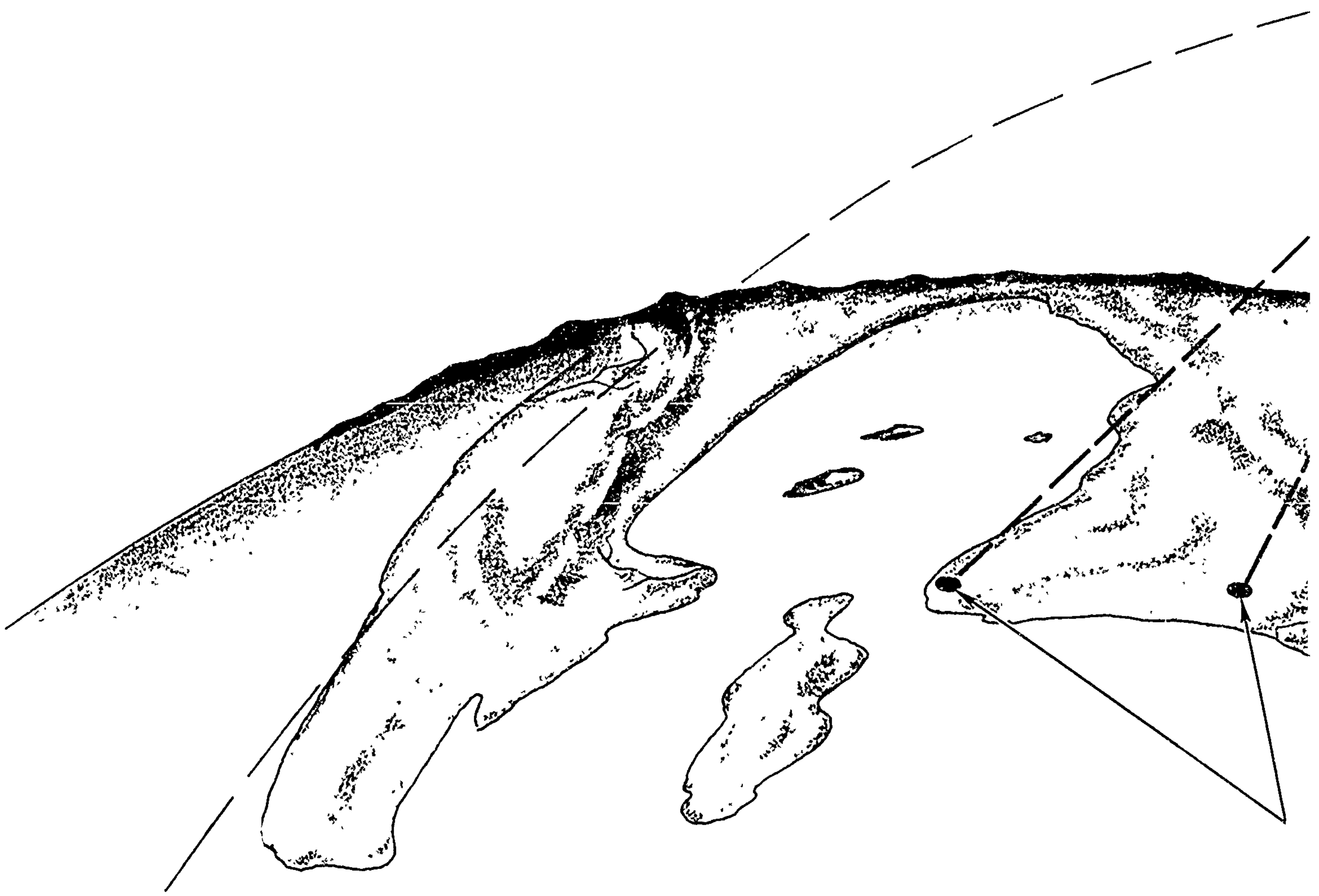
2.3.2 Related Technical Manuals. The following technical manuals have been prepared by Cubic Corporation for the Geodetic SECOR System:

System Description

(1) IM/59-1 Instruction Manual for Geodetic SECOR, Volume I, ~~RF Equipment~~, ~~1 June 1961~~ / January 1961

(2) IM/59-1 Instruction Manual for Geodetic SECOR, Volume II, ~~System Description~~, ~~1 January 1961~~ RF Equipment / June 1961

(3) IM/59-1 Instruction Manual for Geodetic SECOR, Volume III, Data Handling Equipment, 5 September 1961



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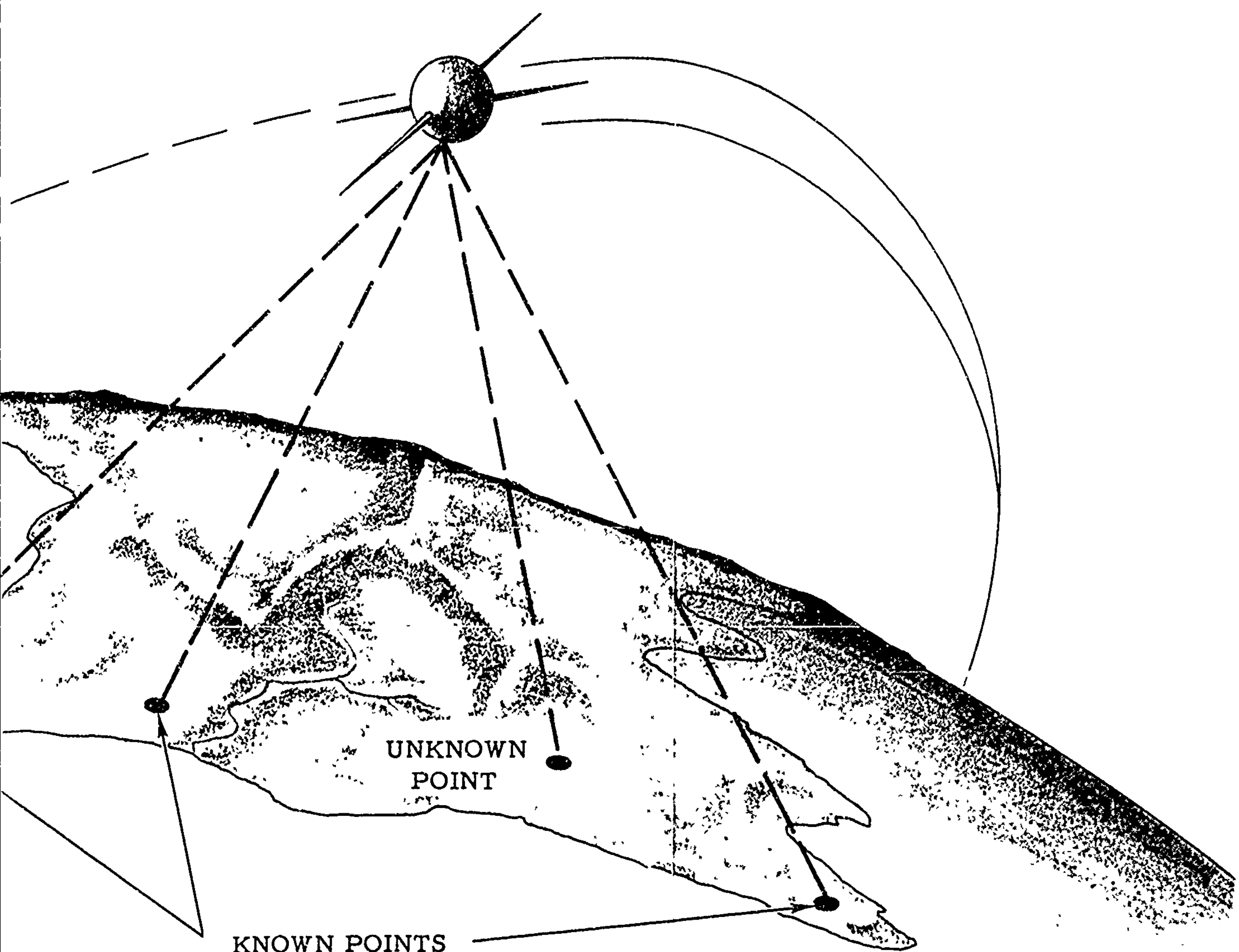


Figure 2-1. Satellite Geodesy

B

2.4 Fulfillment of General System Requirements. In accordance with the contractual requirements Cubic Corporation has delivered three Geodetic Spacecraft (figure 2-2) for use in conjunction with the Geodetic SECOR System. (The Spacecrafts are designated as S/N 1 prototype, S/N 2 flight and S/N 3 flight-backup. The serial numbers are located at the base of the respective instrumentation tubes.) Cubic has designed, fabricated, assembled, and tested each spacecraft for a useful life of at least one year in orbits between 500 and 1200 nautical miles (nm) and at various inclinations. Table I lists the major specifications. The spacecraft design permits injection into orbit by a NASA Scout vehicle or in a pick-a-back configuration aboard a Transit satellite. The spacecraft payload includes a SECOR TR-17 transponder, a miniature solid-state telemeter system for monitoring nine critical parameters, three omnidirectional antenna arrays, and solar cells and storage batteries sufficiently redundant to insure the required operational life. GIMRADA, at the recommendation of Cubic and NASA, has permitted the telemetry transmitter to serve as a beacon for NASA Minitrack sites during the initial tracking and computation of the spacecraft orbit. The original requirement for a separate beacon transmitter was waived.

2.5 Fulfillment of System Design Requirements.

2.5.1 Compatibility with Launch Vehicles. Cubic has provided a spacecraft configuration which is compatible with the NASA Scout launch vehicle, the Transit series of spacecraft, and with the composite I Multiple Satellite. Figure 2-3 shows the position of the Geodetic Spacecraft in the Composite I Multiple Satellite. A mock-up spacecraft was constructed for conducting compatibility tests with the proposed launch vehicle. The Geodetic Spacecraft is not compatible with the USN Caleb vehicle, as the design restrictions would have been too limiting. A spacecraft suitable for launch by the Caleb would have required extensive payload miniaturization at an appreciable increase in cost and time for design and fabrication.

2.5 2 Maximum Size of Spacecraft. The Geodetic Spacecraft, with all antennas folded, has a maximum height of 31.73 inches, and a maximum diameter at its equator of 21.079 inches

2.5.3 Optimum Shape of Spacecraft. The Geodetic Spacecraft is spherical in shape except in those areas provided for the solar cell plaques and the separation fixture.

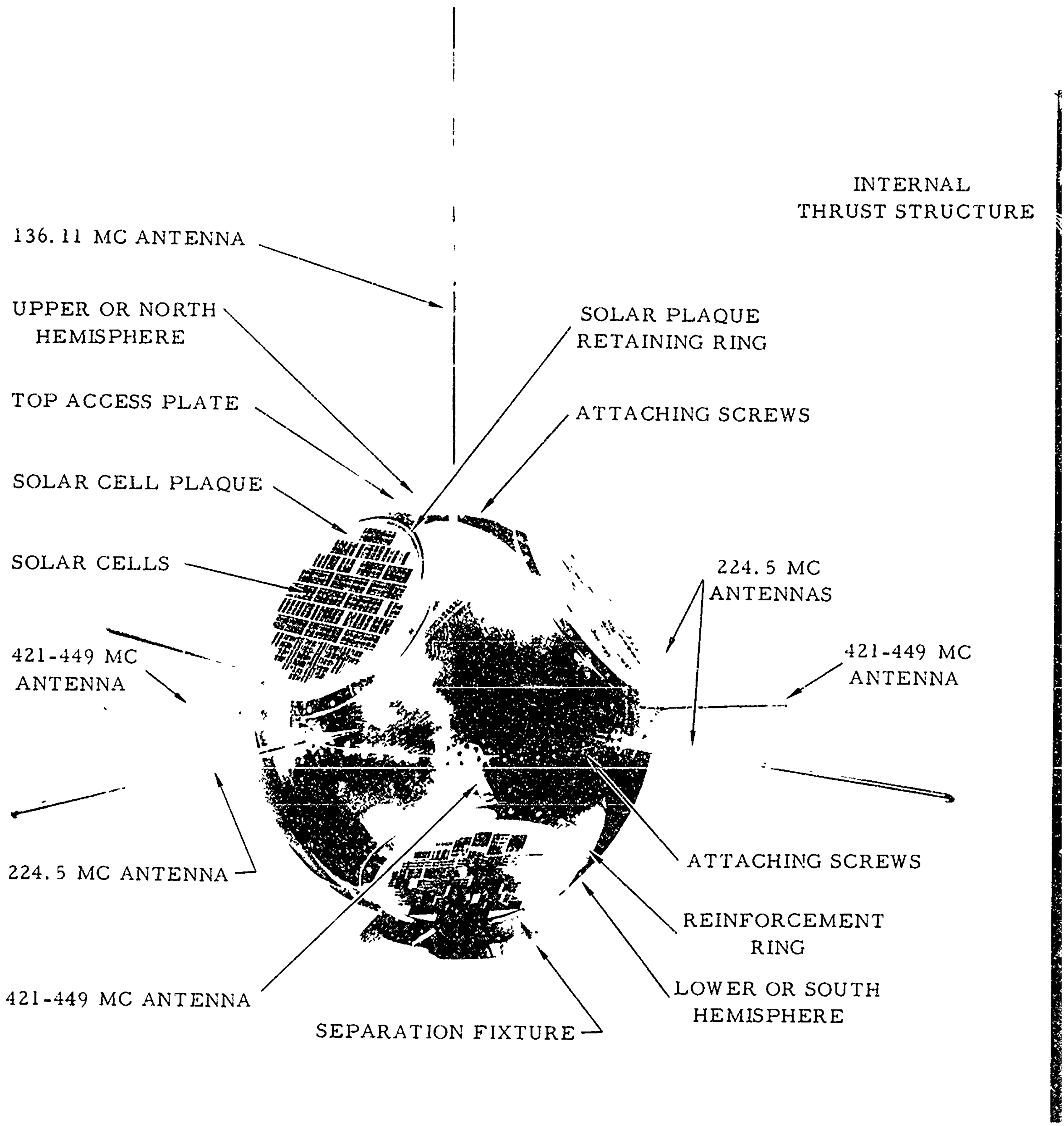
2.5.4 Maximum Weight of Spacecraft. The maximum weight of the Geodetic Spacecraft with all components installed and excluding the vehicle portion of the separation fixture does not exceed approximately 37 pounds.

2.5.5 Lifetime of Spacecraft. Cubic has incorporated suitable redundance of storage batteries, solar cells, and general engineering design to insure a spacecraft lifetime of at least one year. Ground based SECOR transmitters may deactivate the entire electrical system of the Spacecraft (except for the command receiver) at any time if this deactivation is required.

2.5.6 Orbit and Launch Parameters. Extensive environmental testing has shown that the Geodetic Spacecraft will operate as planned when launched and injected into orbit by the NASA Scout or the Thor Able Star type rockets. The solar-cell plaque configuration will provide the required power when the Spacecraft is oriented at any of the various inclinations between 28° and 68° in circular orbits from 500 to 1200 nautical miles.

2.5.7 Thermal Design of Spacecraft. The Geodetic Spacecraft will maintain an interior temperature of $20 \pm 15^{\circ}C$ or better under orbital conditions. The Physics Research Laboratory, U. S. Army Engineer Research and Development Laboratories confirmed Cubic's thermal design and applied a coating of silicon monoxide (SiO) to each spacecraft shell as a government-furnished service. The specified α/ϵ (=1.03) of the spacecraft has been verified by tests performed at the Applied Physics Laboratory of the Johns Hopkins University at Silver Springs, Maryland. The spacecraft telemetry subsystem monitors the shell temperature, solar cell temperature, and battery temperature as required.

2.5.8 Separation Mechanism for the Spacecraft. Cubic has provided each Geodetic Spacecraft with a separation fixture which is compatible with the separation mechanisms used on the NASA Scout vehicle and the Transit spacecraft.



EXTERNAL STRUCTURE

INSTRUMEI
SUPPORT

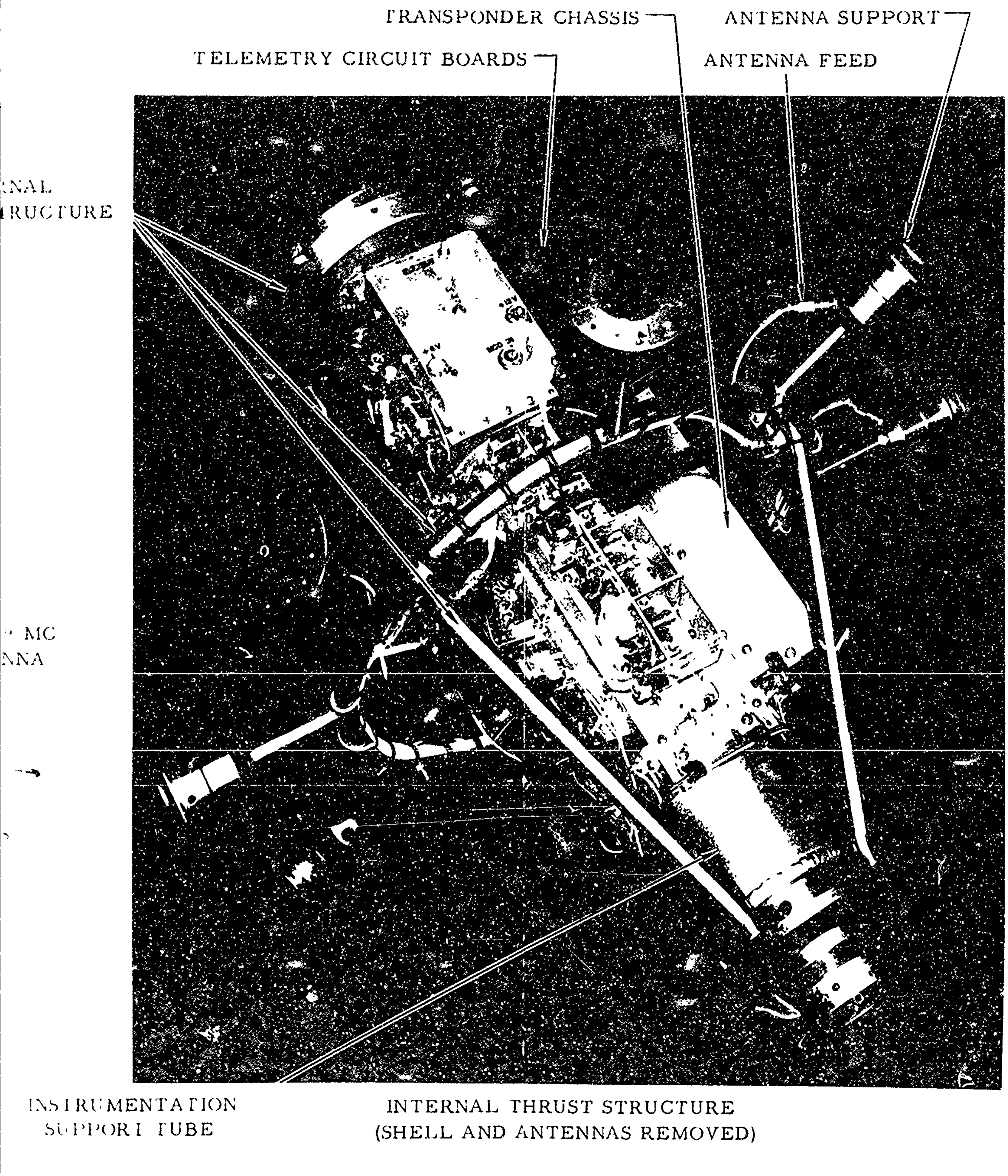


Figure 2-2. Typical Geodetic Spacecraft Construction Details and Component Location

TABLE I
MAJOR SPACECRAFT SPECIFICATIONS

GENERAL	
Lifetime -	One year
Orbital parameters -	Circular orbits of 500 to 1200 nautical miles at various angles of inclination from 28° to 68°
Launch vehicles -	Compatible for launch by NASA Scout, pick-a-back aboard a Transit type spacecraft, or in Composite I Multiple Satellite aboard Thor Able Star vehicle
Ground support equipment -	Cubic test panel, test transmitter, and telemetry test receiver
Stabilized interior temperature -	$20 \pm 15^{\circ}C$
α/ϵ of spacecraft with SiO coating -	1.03
Center of gravity -	On Z-axis 3/4-inch below equator
Moment of inertia (typical) -	0.287 slug ft^2
Static balance -	Within less than 0.005 inch of center of gravity
Electrical access for checkout -	Flight plug located on lower hemisphere
Size of sphere	20 inches diameter
Over-all size	
Antennas folded -	31.73 x 21.079 inches

TABLE I (Cont)

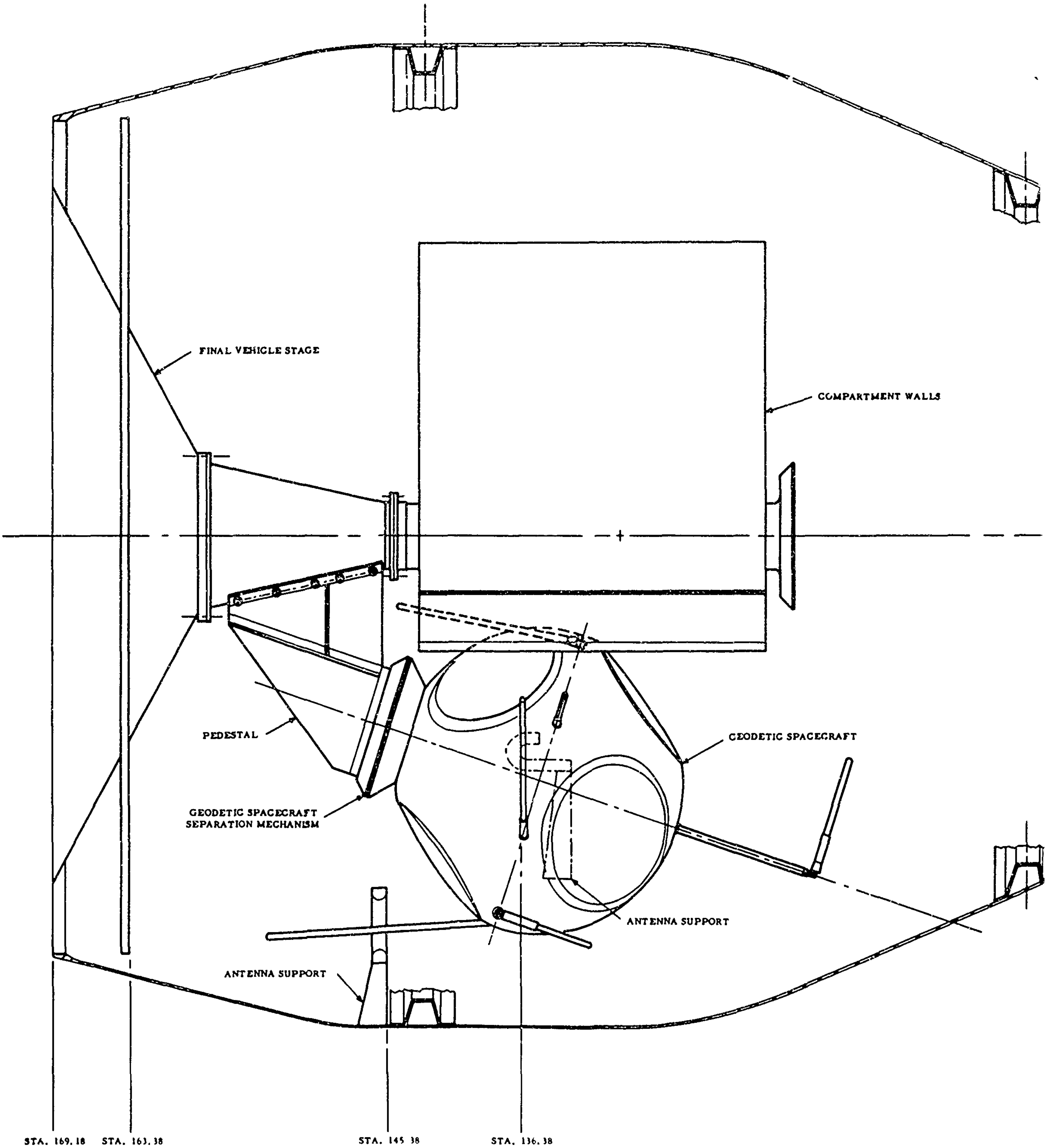
GENERAL (Cont)	
Antennas extended -	42.12 x 52 inches
Weight -	Approximately 37 pounds maximum
SUBSYSTEMS	
External structure -	Consists of internal thrust structure, two aluminum hemispheres, separation fitting, and top access plate
Transponder -	SECOR TR-17 transponder or SECOR TR-27 transponder
Frequencies -	Interrogate - 420.9375 Respond - 449.0 mc - 224.5 mc
Power output -	1 watt minimum at 449.0 mc 1 watt minimum at 224.5 mc
Operation period -	Standby - continuous Transponding - approximately 45 to 60 minutes of each 24-hour period
Telemeter -	Eight-channel FM/PM consisting of nine sensing elements and four printed circuit boards including sensor, timing, commutator, VCO, and transmitter
IRIG band No. -	3

TABLE I (Cont)

SUBSYSTEMS (Cont)	
Frequency -	136. 11 mc
Power output -	100 mw
Parameters measured -	Low calibration, high calibration, battery voltage, transponder plate current, transponder plate voltage, solar cell temperature, battery temperature, shell temperature, and separation signal
Power Supply	
Encapsulated battery package -	Contains two parallel strings of 10 Sonotone Nicad type WS-103 battery cells connected in series
Six solar cell plaques -	Each plaque contains 160 cells arranged in five strings with over-all conversion efficiency of 12 per cent
Power output -	28 ma at 12V to SECOR TR-17 transponder during standby operation 50 ma at 12V to telemetry system during operation 2. 2 amperes at 12V to SECOR TR-17 transponder during transponding operation

TABLE I (Cont)

SUBSYSTEMS (Cont)	
Antennas (omnidirectional) -	Four dipole turnstile for 224.5 mc Four dipole turnstile for 421-449 mc One dipole for 136.11 mc. (All antennas may be folded to facilitate compatibility with launch vehicle heat shield.)
Instrumentation support tube -	Attached to poles of internal thrust structure: (provides support for electronics chassis and serves as container for battery package and TR-17 power converter).



FINAL VEHICLE STAGE
COMPARTMENT WALLS
PEDESTAL
GEODETIC SPACECRAFT
GEODETIC SPACECRAFT SEPARATION MECHANISM
ANTENNA SUPPORT
ANTENNA SUPPORT
STA. 169.18
STA. 163.38
STA. 145.38
STA. 136.38

A

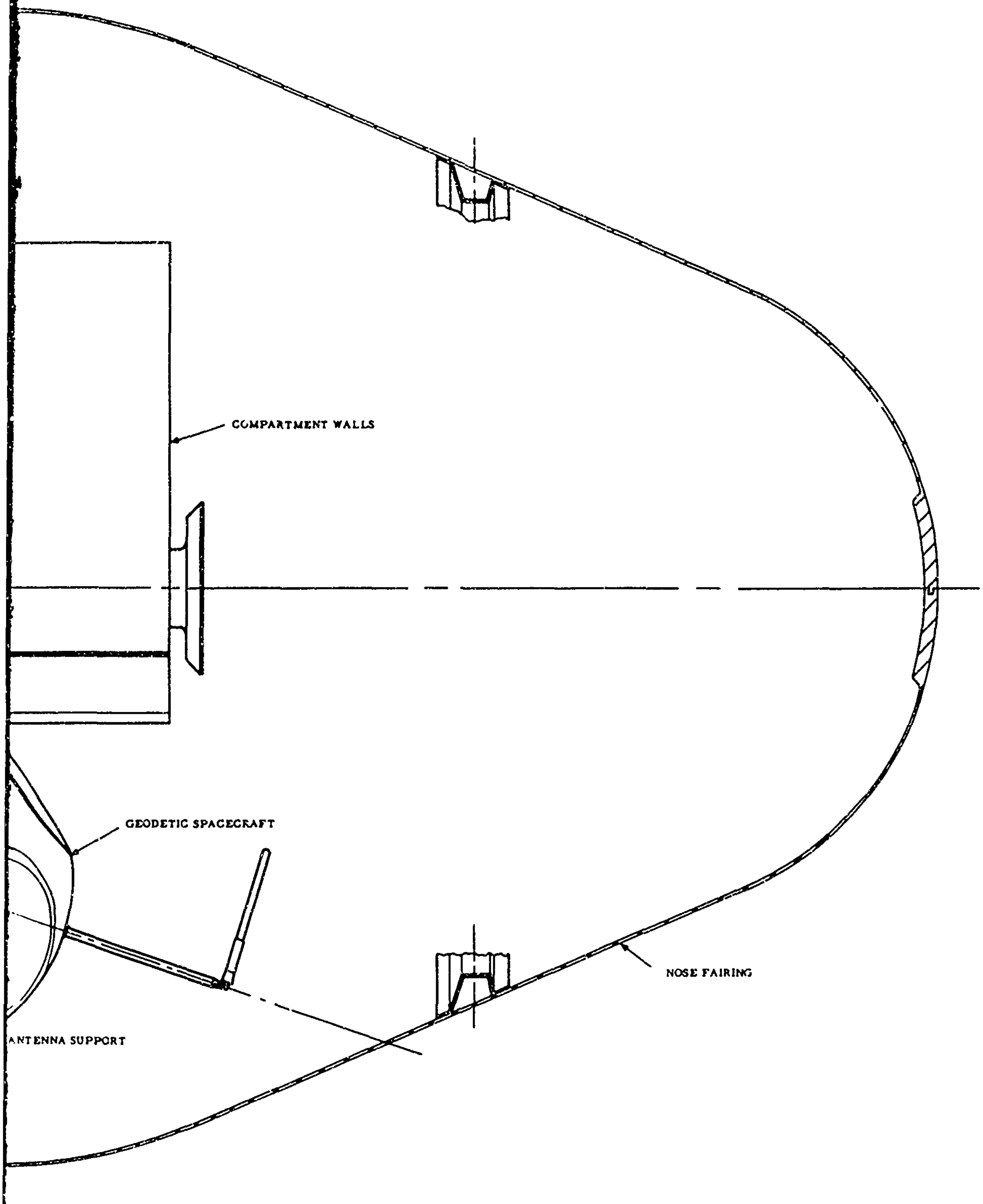


Figure 2-3. Geodetic Spacecraft in Composite I-Multiple Satellite

2.5.9 Interference Suppression. Cubic has designed the SECOR TR-17 transmitter and the telemetry transmitter with the necessary electrical shielding and filtering to provide operation as an interference-free system. The spacecraft generates no mechanical, electrical, or rf interference with the vehicle command, telemetry, or radio tracking and guidance systems. The spacecraft electronics will function properly when integrated with the vehicle system during flight conditions.

2.5.10 Integration with Launch Vehicle. Each Geodetic Spacecraft is a complete and integral system. Cubic is providing all services required to integrate the Spacecraft with the proposed launch vehicle.

2.5.11 Ground Support Equipment and Services. Cubic is providing the test panel, a test transmitter (ground station simulator), a telemetry test receiver, and the associated test equipment to insure adequate instrumentation during both the test and the launch phases of the program. Cubic has supplied a sling for lifting the spacecraft into position on the satellite cluster. Cubic-furnished services include the delivery, checkout, and installation of the Spacecraft on the launch vehicle and the monitoring of flight readiness and performance. Tracking and the processing of flight data is the responsibility of NASA and is coordinated directly by the Contracting Officer.

2.6 Fulfillment of Subsystem Design Requirements. The following paragraphs describe the six subsystems designed and fabricated by Cubic for each of the Geodetic Spacecraft; included are those subsystems required by contractural specifications.

2.6.1 External Structure Subsystem. The external structure of each of the Spacecraft (manufactured by Brooks and Perkins, Inc. of Detroit, Michigan) consists of an internal thrust structure, two aluminum hemispheric shells, a top access plate, and a separation fixture. The internal thrust structure is a welded magnesium frame which supports the electronic instrumentation tube, nine antennas, the separation fixture, and the two hemispheres of the shell. Each hemisphere of the shell supports three solar cell

plaques, and provides the thermal stabilization for the interior of the Spacecraft. The separation fixture is a machined aluminum-alloy flange which mates with a similar fitting on the last stage of the proposed launch vehicle.

2.6.2 Transponder Subsystem. The principle function of the Geodetic Spacecraft is to carry and power a GFE SECOR TR-17 or TR-27 transponder. The transponder subsystem, therefore, consists of a modified TR-17 or TR-27 transponder and the interconnecting wiring. As the original physical configuration of the transponder (packaged in three canisters) was incompatible with the Spacecraft, the six electronics chassis were remounted on and inside the instrumentation support tube; this resulted in a considerable reduction in weight. Cubic has modified the circuitry to include two telemetry-call tone filters and amplifiers for turning the telemetry on and off at ground command. The additional amplifiers tuned to 570 mc and 575 mc permit the termination of all telemetry transmissions at any time if required. As the coherent-carrier mode of operation is not required for geodetic SECOR purposes, the related circuitry has been removed from the transponder, thereby enhancing reliability. The transponder receives interrogation signals at 421 mc from DME ground stations and transfers the modulation frequencies on the interrogation signal to locally generated offset carriers. The offset carriers, 449.0 mc and 224.5 mc, return the interrogation signal modulations to the DME station. The transponder operates continuously in standby condition as a receiver until a select call transmission is made by a DME station. The select call signal transfers the transponder to an operate condition. The transponder operating (transponding) period, based on the power available from solar cell sources, is scheduled to be approximately 45-60 minutes of each 24-hour period.

NOTE

> As the SECOR transponders are GFE and only two TR-17 models were made available for this project, Cubic performed modification and reassembly on the two units furnished. These transponders were installed in the flight and flight-backup models. The transponder was

removed from the flight-backup model and reinstalled in the prototype. The prototype is now at NRL for use in the launch integration test. A TR-27 transponder was made available on 29 September and will be installed in Spacecraft No. 3. The TR-27 transponder does not include the coherent-carrier circuitry.

2.6.3 Telemetry Subsystem. The miniature solid-state FM/PM telemetry subsystem consists of eight sensing elements, an eight-channel commutator, a multiplexer, a voltage-controlled oscillator (VCO) operating on IRIG band No. 3, and a transmitter operating at 136.11 mc at 100 mw. Parameters selected for measurement include:

(1) 2V calibration

(2) 4V calibration (full scale)

(3) Battery voltage

(4) SECOR TR-17 transponder plate current

(5) SECOR TR-17 transponder plate voltage

(6) Solar cell temperature

(7) Battery temperature

(8) Shell temperature

(9) Separation signal. (The non-separation signal appears as a spike in the battery voltage measurement.)

The transmitted signals are compatible with NASA Minitrack installations, and the geodetic SECOR ground-based transmitters control the telemetry on-off power function. Connections from the commutator to a flight plug located on the lower shell permit monitoring of the telemetry and information channels during prelaunch checkout.

2.6.4 Power Supply Subsytem. The power supply subsystem contains the storage battery package, six solar cell plaques and the charging network. The battery package consists of two parallel strings of 10 Sonotone Nicad type WS-103 battery cells connected in series. The cells are arranged in a support assembly in groups of four in a square array stacked five high. The entire assembly is potted in Eccosil 4640 to provide a pressure seal. Each solar cell plaque contains 160 Hoffman Electronics type 120C G or equivalent cells arranged in five strings. Each cell is provided with an Ocli type 207Scc450-1 filter with UV and AR coatings. The cells have been carefully selected and calibrated to insure an overall conversion efficiency of 12 per cent. Isolation diodes are incorporated in each string to prevent short-circuiting the power supply in the event of failure of an individual string or cell as a result of factors such as meteorite damage. The subsystem provides a continuous 28 ma at 12V for the TR-17 transponder during standby, and 2.2 amperes at 12V during operation. During sunlit hours, the susbystem will replace all the power consumed by the Spacecraft during the entire orbit. A flight plug mounted on the lower shell of the Spacecraft permits external control of the on-off switching of power to all electronic components during prelaunch checkout operations. Battery connections at the flight plug permit charging from external sources. The transponder power converter chassis is mounted on the potted battery package.

2.6.5 Antenna Subsystem. The omnidirectional antenna subsytem consists of two independent SECOR arrays and a single telemetry dipole. Each SECOR array is a four-dipole turnstile with the individual elements located at 90 degree intervals around the equator of the spacecraft. A separate array is required for the high (421-449 mc) and the low (224.5 mc) frequencies. The telemetry antenna is located at the north pole of the Spacecraft. All antennas are folded during launch and automatically extended to their proper operating length upon separation of the Spacecraft from the launch vehicle.

2.6.6 Instrumentation Support Tube Subsystem. The instrumentation support tube, permanently attached to the poles of the internal thrust structure, provides the chassis for mounting the

SECOR transponder and the telemeter, and serves as a container for the battery package and the power converter.

2.6.7 Despin Mechanism. As the proposed final launch stage is not spun, a despin mechanism is not required.

2.7 Fulfillment of Inspection and Test Requirements.

2.7.1 Component Inspection and Tests. Cubic has performed the inspections required to insure compliance with the standards and specifications for the Spacecraft. The necessary subsystem tests have been conducted to determine that the reliability and specifications of each component are satisfactory for their intended use.

2.7.2 System Inspection and Tests. Inspections and tests have been made to determine that the system meets contractural requirements. Each Geodetic Spacecraft was balanced statically and dynamically, and the α/ϵ of the spacecraft was verified. The following environmental tests have been conducted sucessfully:

(1) Vibration tests - Random and sinusoidal vibration tests at levels at least 1.0 times the expected flight levels

(2) Acceleration tests - Acceleration tests to 35g in the direction of launch thrust with 1g in a transverse axis

(3) Thermal-Vacuum tests - Thermal-vacuum tests at 1×10^{-5} mm Hg at temperatures expected in flight, and at temperatures at least 10 $^{\circ}$C above and below the predicted extremes for the specified duration.

SECTION III
INVESTIGATION

3.1 Scope of Section. This section contains the factual data for all the work accomplished on the Geodetic Spacecraft Project during the period from 28 February 1961 to 30 October 1961. It covers the design fabrication, assembly, and testing of the three Spacecraft and of each of their functional subsystems. Detail data includes all pertinent tests and test results, both positive and negative; lists of all standard instruments employed in testing and measurement; a description of all difficulties encountered; derivations of all mathematical formulas developed which were unique to the problem; and supplementary photographs, tables, charts, and curves as necessary for lucid presentation.

3.2 General System Problem. The general system problem is to design, fabricate, assemble, and test three Geodetic Spacecraft. The Spacecraft must be completely integrated and will consist of a mechanical thrust structure to support the SECOR TR-17 transponder, storage batteries, a small solid-state telemeter, an antenna subsystem, and the outer shell The outer shell must support a solar cell network and be surface-finished with a temperature-stabilizing coating of silicon monoxide (SiO). The spacecraft must be able to be launched by a Scout vehicle, or in pick-a-back configuration aboard a Transit spacecraft, or in a satellite cluster aboard a Thor Able Star rocket.

3.3 General Philosophy of System Development. The urgency of the SECOR program required that the spacecraft be as simple and reliable as possible, and that proven satellite techniques, and existing tooling and hardware be utilized to the maximum. In view of these requirements, a spacecraft was designed which combined a mechanical structure and thermal-stabilization similar to that used by Vanguard II, and solar cell plaques and telemetry based on the NRL Lofti configurations. Cubic designed the antenna subsystem and the storage battery power supply, and based the environmental test program on the requirements established for the Transit series of spacecraft. The design, fabrication and assembly of the three Geodetic Spacecraft were accomplished on a subsystem basis beginning with the mechanical structure, and proceeding through the transponder subsystem, the telemetry subsystem, the power

supply subsystem, the antenna subsystem, the instrumentation support tube, thermal stabilization, system integration, dynamic balancing, and system testing. The following discussion of the complete history of this development is presented in the above sequence.

3.4 Developmental History of External Structure Subsystem. Cubic Corporation, as the prime contractor, selected the basic mechanical design for the external structure of the Spacecraft. Brooks and Perkins, Inc. of Detroit, Michigan, fabricated and assembled the mechanical structures.

3.4.1 General Problem. In order to fulfill the general system requirements and to meet the established project dates, an immediately available mechanical structure was required which would support the SECOR transponder and its related electronics under the proposed launch and orbital conditions.

3.4.2 Preliminary Considerations. The preparation of a detail specification and the investigation to determine qualified manufacturers required a detailed preliminary analysis of the design requirements. The urgency of the SECOR program required that the basic mechanical design be as simple and reliable as possible, and that maximum use be made of proven techniques and available tooling and hardware. Payload characteristics, the type of launch vehicle (NASA Scout or Transit pick-a-back), and the proposed orbital parameters further influenced the mechanical design. The SECOR TR-17 transponder, its power supply, and related electronics including solar cells and storage batteries limited the possible minimum volume of the Spacecraft and established the required internal environment. The launch vehicles under consideration restricted the total size and weight of the Spacecraft and determined the thrust parameters of the internal structure.

3.4.3 Development Details. After an extensive study of the design requirements Cubic selected a mechanical structure similar to that used for the Vanguard II spacecraft. (See figure 3-1.) This mechanical structure consists of the separation fixture for attaching the Spacecraft to the launching vehicle, the internal thrust structure for supporting the electronic equipment and the outer shell of the Spacecraft. The following paragraphs describe the components of the mechanical structure and list their major parameters.

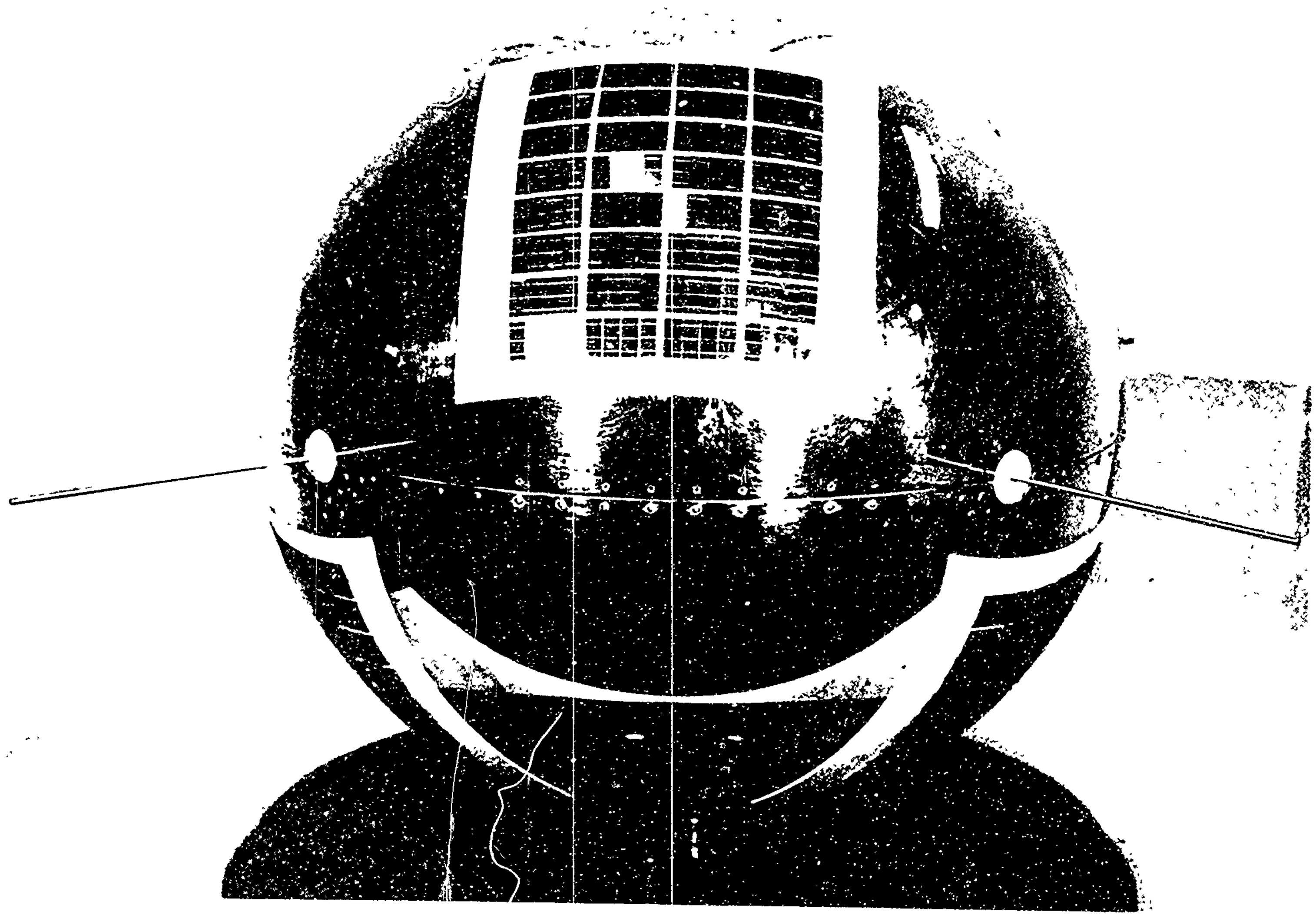
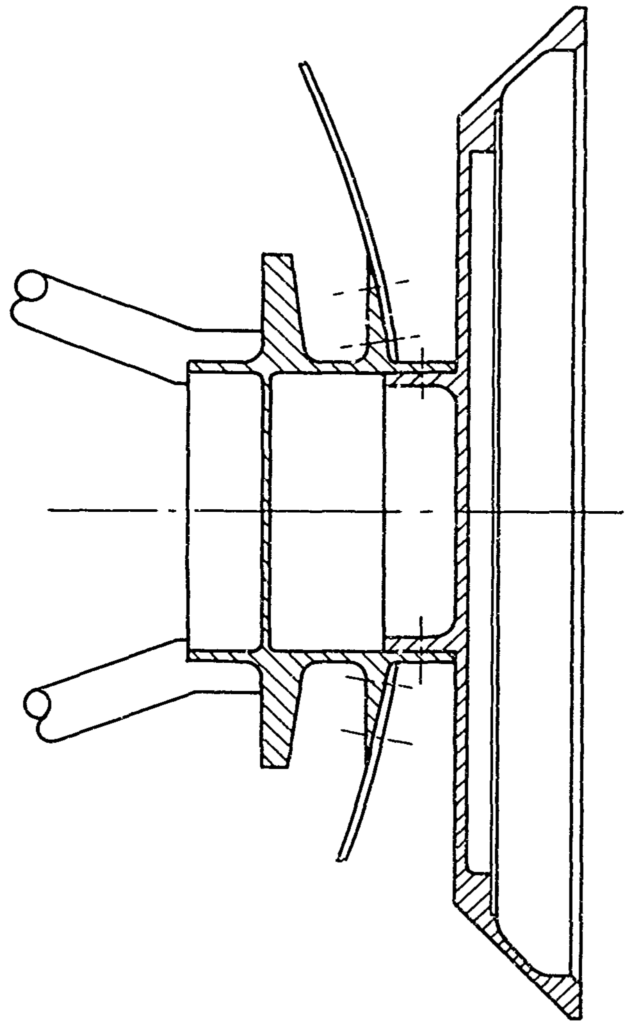


Figure 3-1. Vanguard II Type Spacecraft

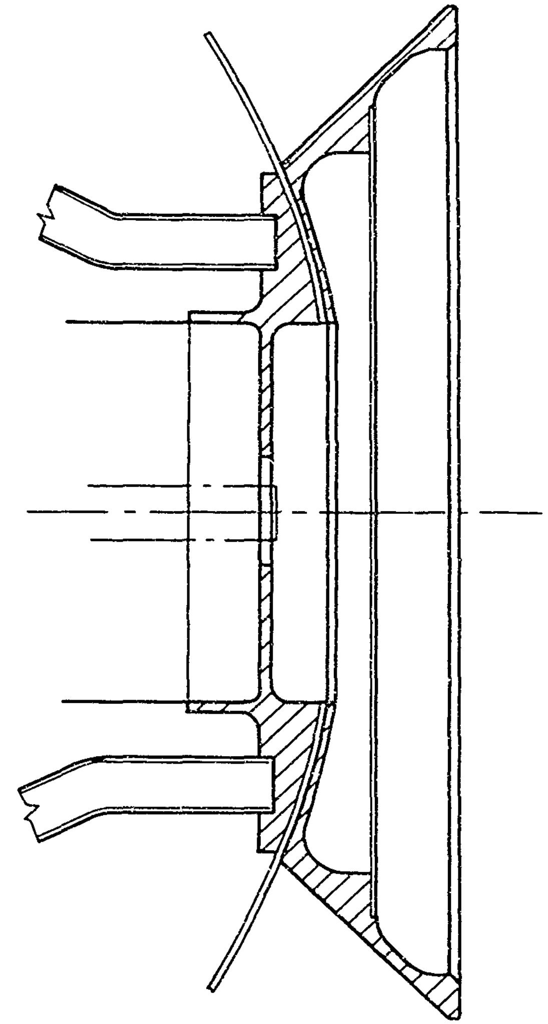
3.4.3.1 Development of the Separation Fixture. The separation fixture is a machined aluminum alloy (6061-T6, QQ-A-327) flange which is secured through the shell to the south pole of the thrust structure. The flange mates with a similar fitting which is attached to the last stage of the launch vehicle or to the top of the Transit spacecraft. The mated fittings enclose a spring or other separation device, and an explosive retaining clamp secures the assembly. The separation fixture on Spacecrafts No. 2 and No. 3 differs from the one utilized on prototype Spacecraft No. 1. Data collected during vibration testing and the finalization of the launch vehicle resulted in modification of the fitting. The modified fitting was fabricated after Spacecraft No. 1 had been completed. Figure 3-2 provides a comparison of the two structures. The separation fixture used on the flight models is surface-polished to ~~0.063 inch,~~ less than four (4) micro inches and coated with silicon monoxide (SiO) for thermal stabilization. Twenty-four stainless steel screws secure the fixture to the Spacecraft.

3.4.3.2 Development of the Internal Thrust Structure. The internal thrust structure is a welded magnesium frame which supports the electronic equipment, the separation fixture, the nine antennas, and the two hemispheres of the spacecraft outer skin. (See figure 3-3.) The four vertical tubes and the equatorial tube are magnesium (0.50 inch OD x 0.35 inch wall). The thrust structure at the south pole is machined from magnesium alloy, and the ring at the north pole is a weldment of magnesium alloy. The entire structure is stress-relieved after welding. The prototype frame differs slightly from the flight models because of the type of separation fittings employed.

3.4.3.3 Development of the Shell. The outer shell of the Spacecraft consists of two hemispherical sections which cover the internal electronics, provide thermal stabilization, and support the solar plate assemblies. The shell has an outside diameter of 20 inches and is fabricated of spun aluminum. The exterior surface is machined and polished to provide a smooth surface suitable for a silicon monoxide thermal coating. The top half of the shell is 0.030 inch thick and is attached to the bottom half, the antenna supports, and the upper thrust ring with stainless steel screws. It is easily removable for access to the interior of the Spacecraft. The bottom hemisphere is 0.030 inch thick except in the area where the separation fitting attaches (0.050 inch thick). The eight splice channels riveted



PROTOTYPE SEPARATION FIXTURE



MODIFIED SEPARATION FIXTURE

Figure 3-2. Separation Fixtures

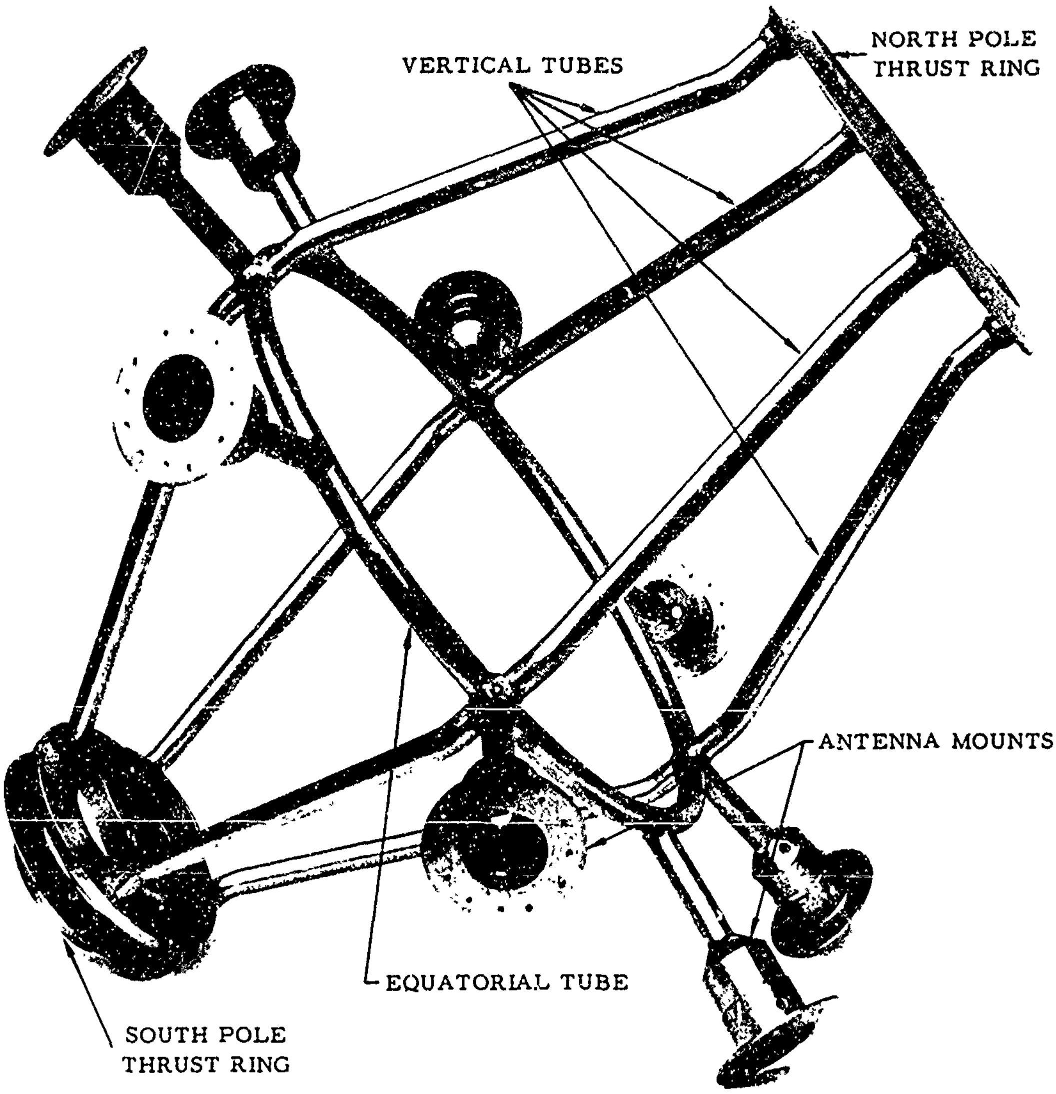


Figure 3-3. Internal Thrust Structure, Prototype

around the equator are tapped to receive the attaching screws of the upper hemisphere. The removable plate at the north pole of the Spacecraft is held in place by screws which secure it to the internal thrust structure.

3.4.3.3.1 Each of the six 9.065-inch circular openings in the shell of the Spacecraft supports a solar cell plate. The openings in the upper half of the shell are located on projected azimuth angles of 120 $\pm 1^{\circ}$ from the center of the sphere. The three openings on the lower half of the shell are displaced from those in the upper half by 60 degrees in projected azimuth angle. Originally the plane of each opening was to be displaced 45 degrees from the equator of the Spacecraft. The lower half of Spacecraft No. 1 is constructed in this manner. The openings in the upper half of the first unit and those in both hemispheres of the flight models are displaced at an angle of 38°. (See figure 3-4.) A threaded reinforcement ring, riveted into each opening, mates with a retaining ring (figure 3-5) which locks the solar plate into position. The retaining rings have an inside diameter of 8.687 inches and fit over a flange on the bottom of the solar plates. The four holes located at 90-degree intervals on the top of the retaining ring mate with a special spanner wrench which is used to secure the ring. The exterior surface of the rings is coated with silicon monoxide (SiO) for thermal stabalization.

3.4.4 Investigation to Determine Qualified Manufacturers. As Cubic had selected the Vanguard II-type mechanical structure, it was determined that the choice of manufacturers was immediately limited to Brooks and Perkins, Inc. of Detroit, Michigan who manufactured the Vanguard II vehicles for the government.

3.4.5 Fabrication of External Structure. On 3 March 1961, Cubic negotiated with Brooks and Perkins, Inc. for the manufacture of the external structure for one Geodetic Spacecraft. Representatives of Brooks and Perkins, Inc. agreed to deliver the unit to Ft. Belvoir for the thermal coating by 3 April 1961. The fabrication and assembly of the internal structure and the shell proceeded on schedule. A delay in the finalization of the separation fitting design caused a consequent delay in the fabrication of the

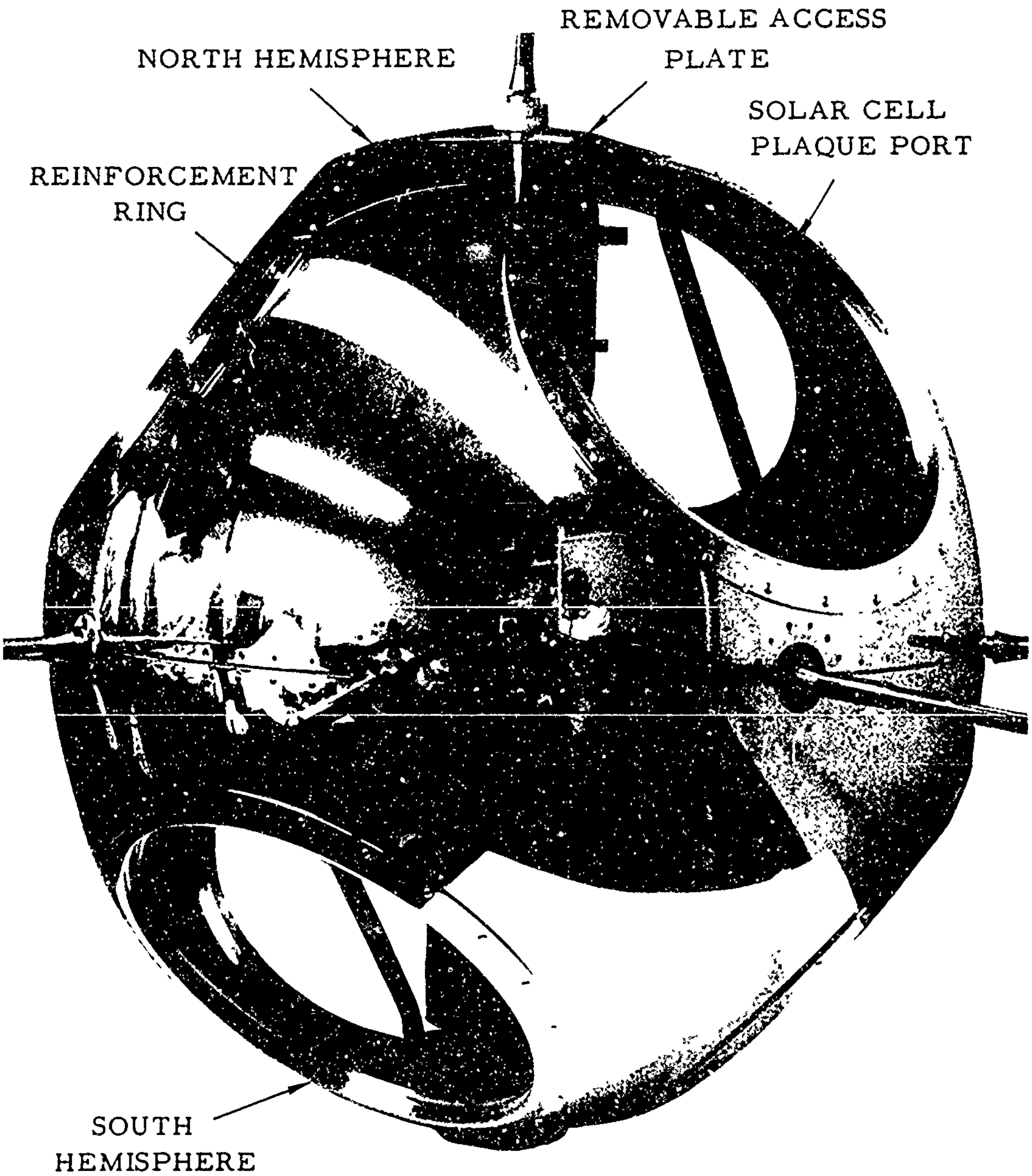


Figure 3-4. Typical Outer Shell

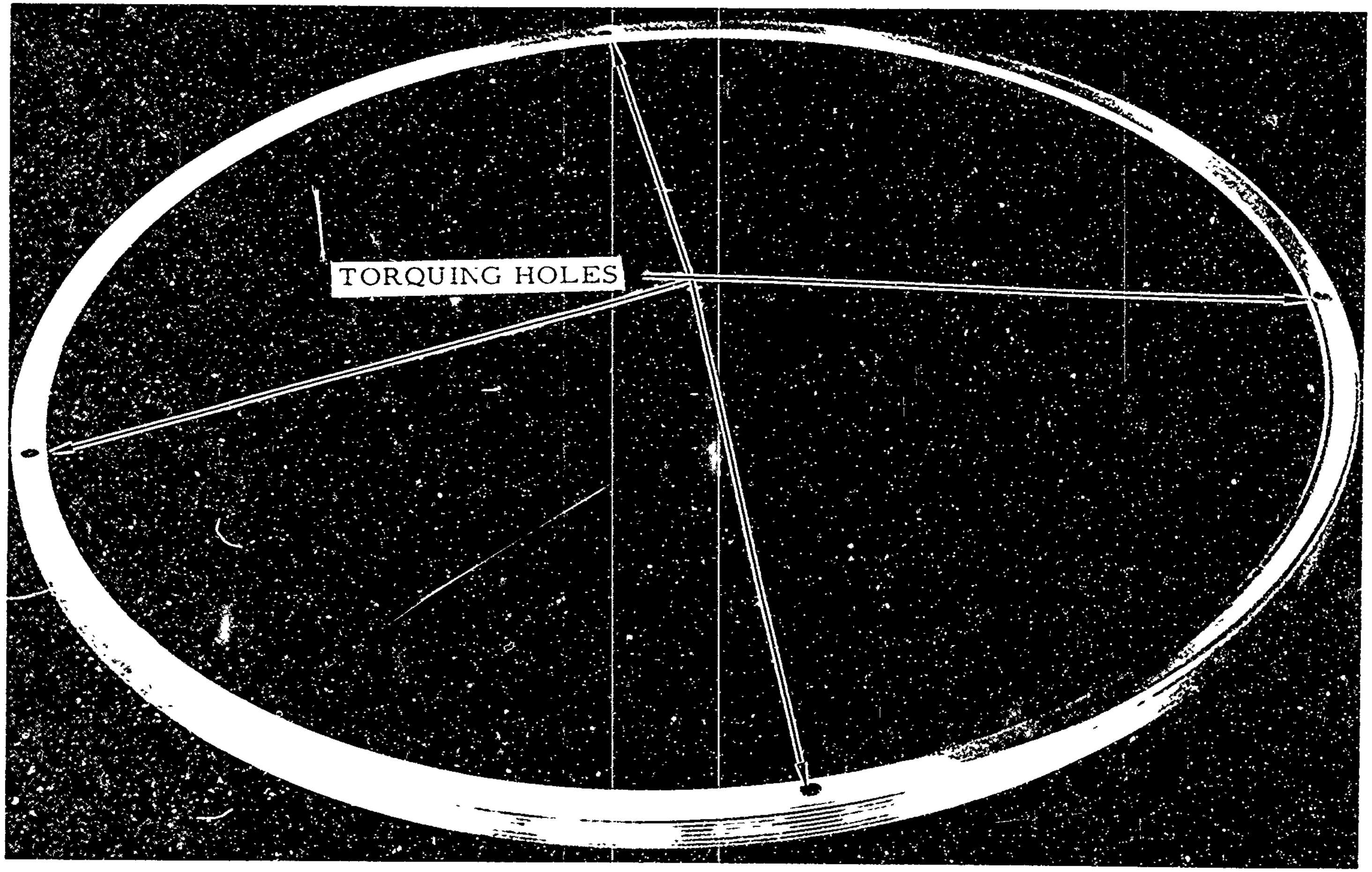


Figure 3-5 Typical Solar Cell Plaque Retaining Ring

Spacecraft. On 28 March the problem was resolved at Ft. Belvior, and the Spacecraft was delivered with a modified Transit fitting.

3.4.5.1 On 3 June 1961, Cubic renegotiated with Brooks and Perkins, Inc. for the purchase of two additional external structures The method of attaching the separation fitting had been changed, and this modification was incorporated into the additional Spacecraft. Relocation of the separation fixture altered the position of the solar plaque openings on the lower hemispheres of the Spacecraft. The plane of each solar plaque on Spacecraft No. 2 and No. 3 is displaced at an angle of 38^{o} from the equator. Originally, the plaque openings were to be located at a 45-degree angle. The only other deviation from the specification occurred in Spacecraft No. 2 in which the splice channels are riveted to the upper half of the shell rather than to the lower half. On the first unit, nuts and screws were used to attach the shell to the antenna mounts. These were modified, and the splice channel was ~~antenna mounts were~~ tapped in order to facilitate assembly and disassembly

3.5 Developmental History of the Transponder Subsystem. The government furnished two SECOR TR-17 transponders and one TR-27 transponder for the Geodetic Spacecraft Project. Cubic, the original manufacturer, modified the TR-17 transponders to make them compatible with the Spacecraft. Prior to installation in the Spacecraft, Cubic performed the tests necessary to insure that the reliability and specifications of each transponder were satisfactory for their intended use.

3.5.1 General Problem. The general problem in the development of the transponder subsystem involved removal of the coherent-carrier mode circuit from the TR-17 models, the addition of telemetry call channels to each unit, and mechanical rearrangement of the chassis.

3.5.2 Preliminary Considerations. In accordance with the system design, Cubic planned to mount all spacecraft electronics on an instrumentation support tube located on the thrust axis and secured to the poles of the Spacecraft. In order to make the GFE transponders compatible with this arrangement, Cubic proposed removing the transponder from its existing canisters (figure 3-6) and remounting the individual chassis on the support tube. With regard

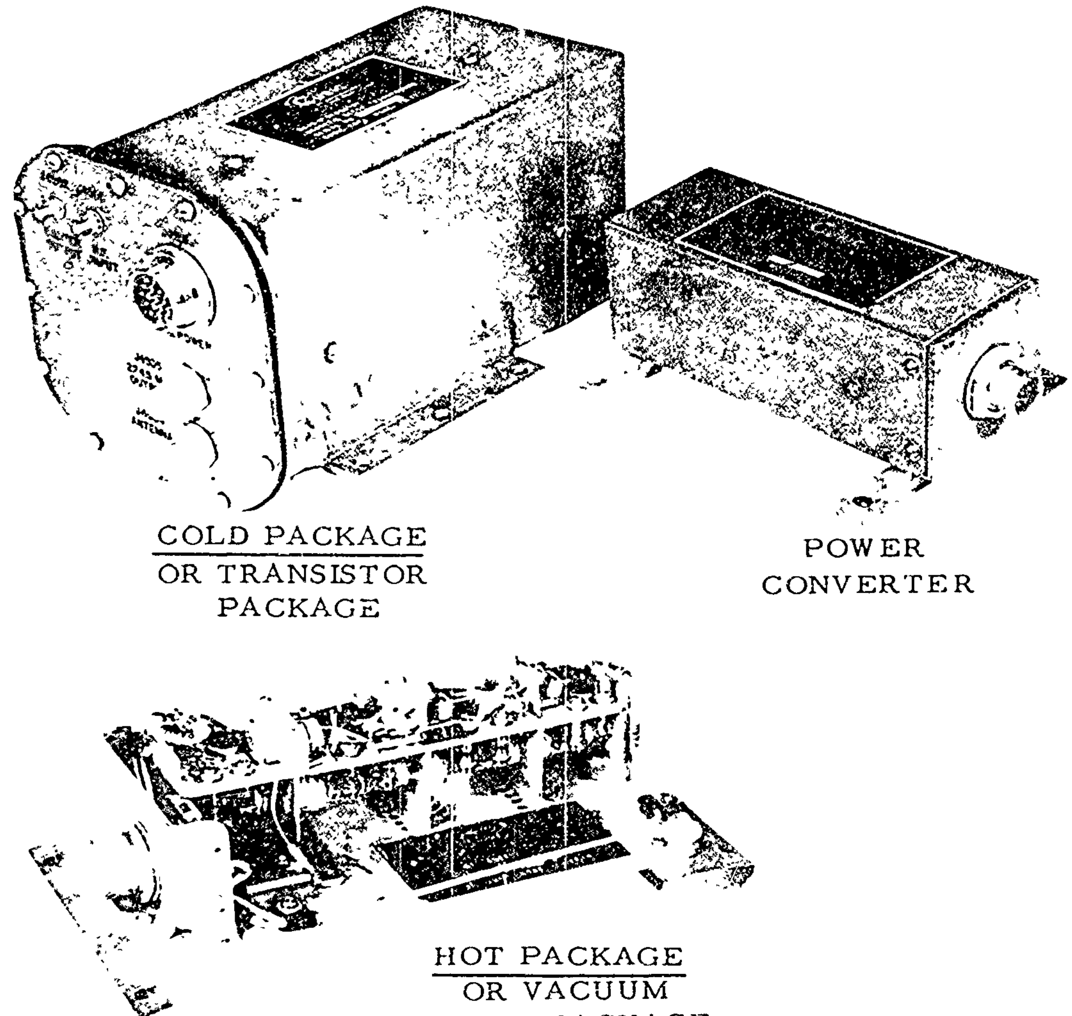


Figure 3-6. SECOR TR-17 Transponder Before Installation in Spacecraft

to the circuitry, the original specification called for a command receiver function to turn the telemeter on and off with a ground-initiated signal. Rather than utilize a separate receiver, Cubic planned to incorporate this function into the receiver circuitry of the transponder. As the coherent-carrier mode of operation is not required for geodetic SECOR purposes, it was to be removed from the TR-17 transponders thereby enhancing reliability.

3.5.3 Development Details. The two GFE TR-17 transponders were disassembled and all but two transistors were removed from the coherent-carrier circuit. Specifically, Q8, Q9, and the cable connecting J4102 and J4006 were removed from the receiver chassis, and Q10, Q11, and crystal Y7 were removed from the coherency-control circuit in the data amplifier. Transistors Q9 and Q12, although part of the coherent-carrier channel, were left in position for purposes of mechanical support. Cubic constructed a 570 mc and a 575 mc tone filter and amplifier for each of the transponders. The amplifiers are identical except for the control crystals. (See figure 3-7.) A two-second tone at 570 ~~mc~~ kc energizes relay K3b, and connects power to the telemetry circuits; tone at 575 ~~mc~~ kc energizes relay K3a which removes power from the telemeter.

3.5.3.1 The following transponder subchassis are mounted on the pentagon sleeve of the instrumentation support tube:

(1) Data amplifier

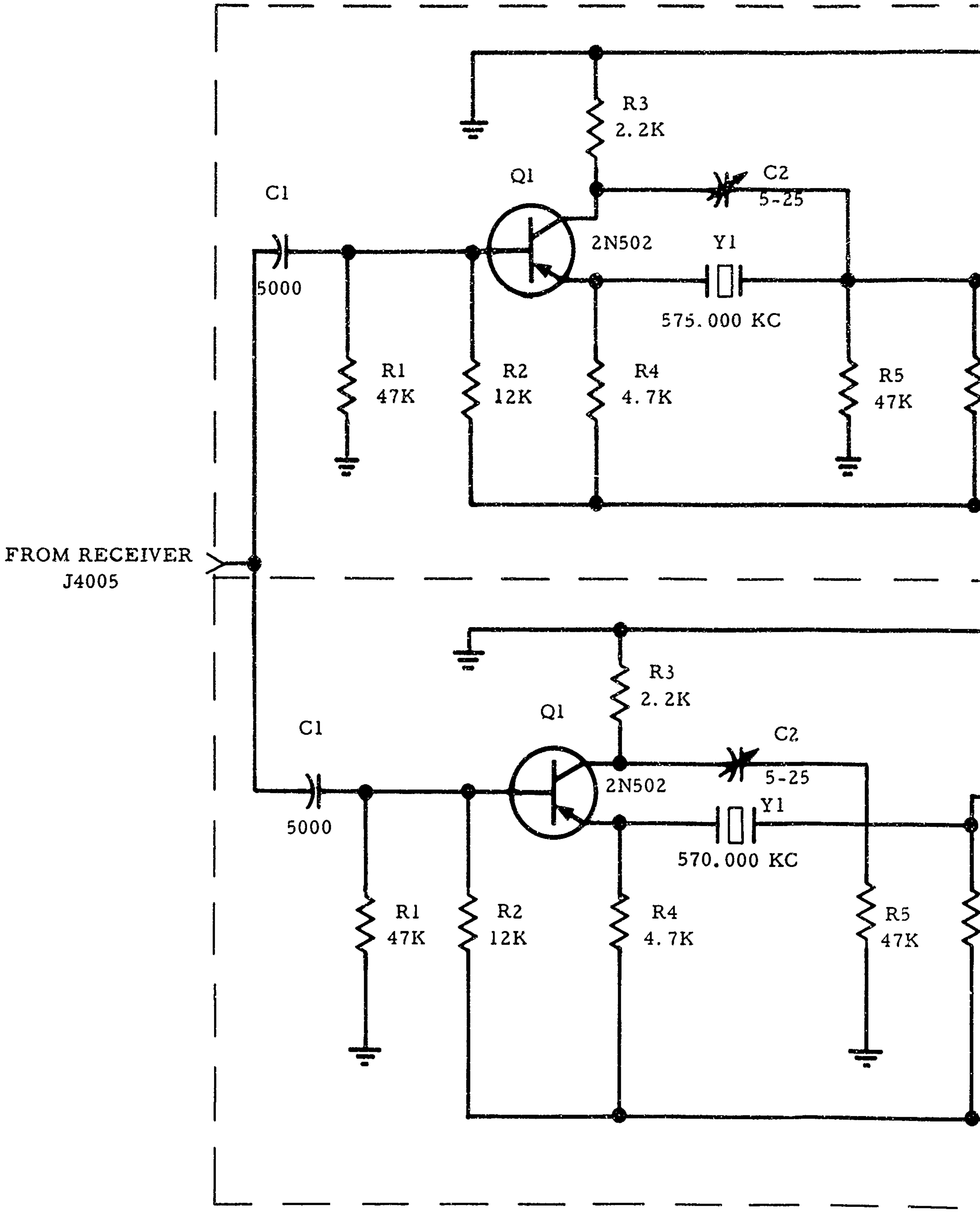
(2) Receiver

(3) Coherent carrier - correlation detector

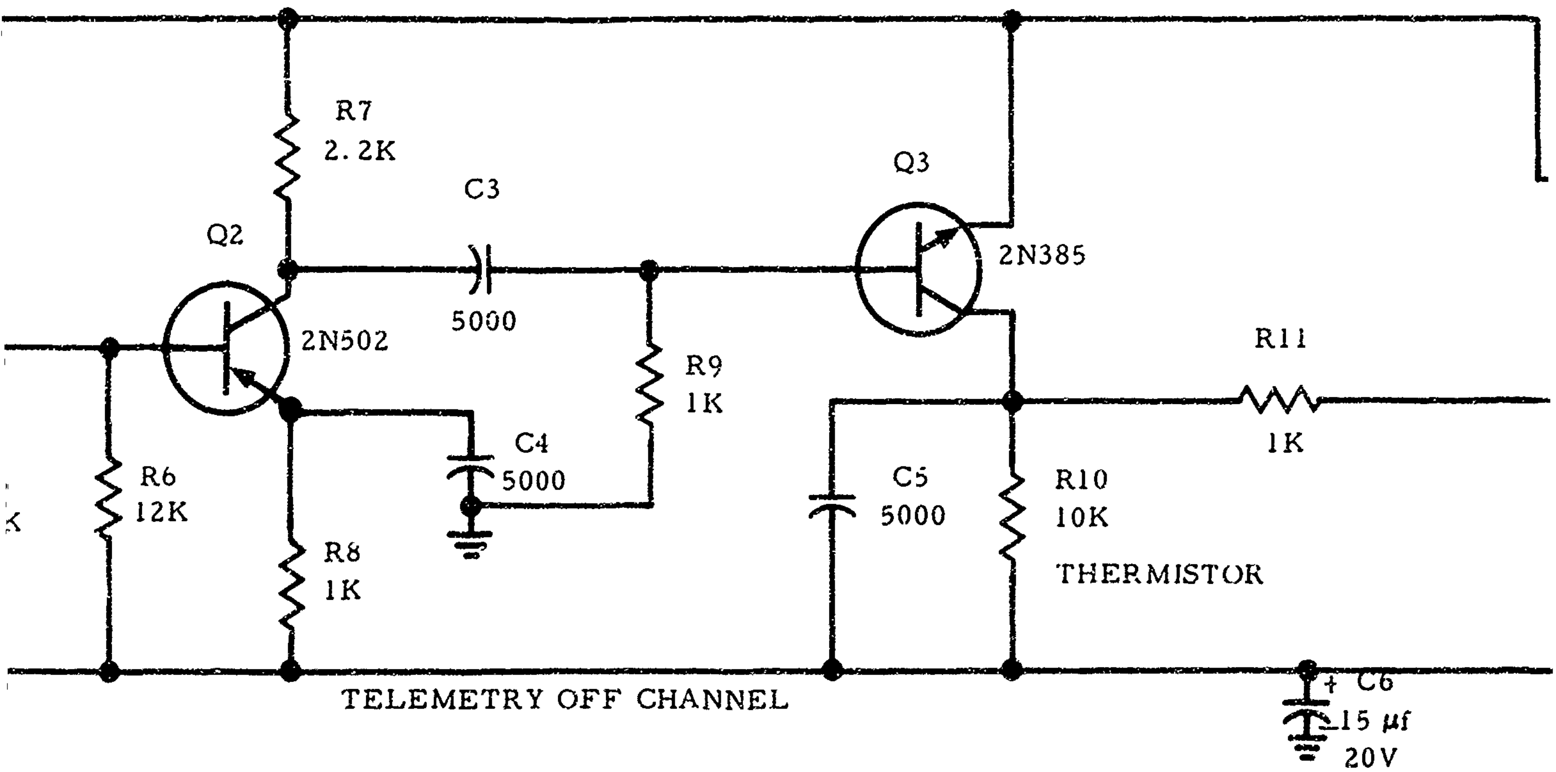
(4) Transmitter

(5) Multiplier.

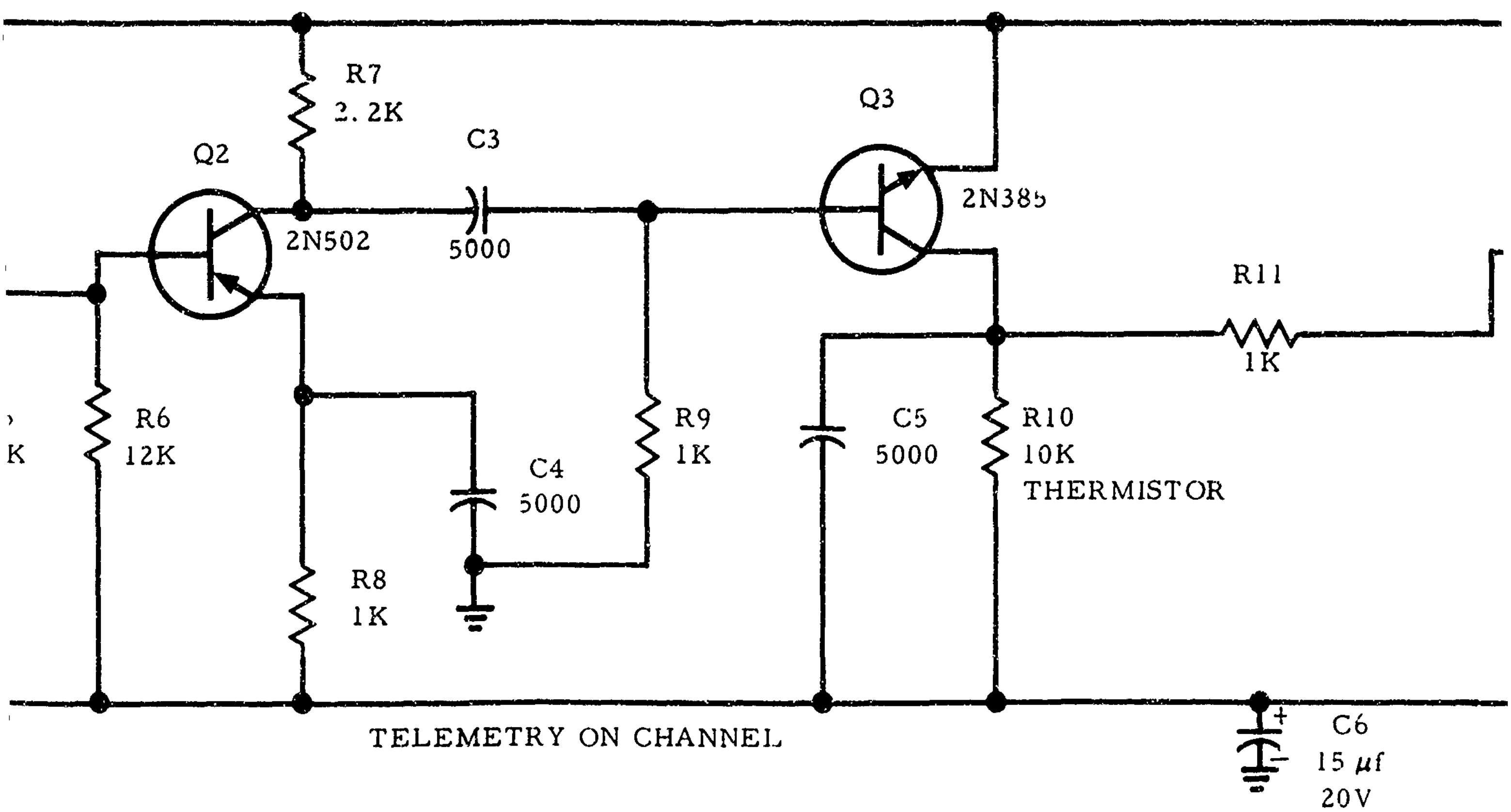
The components of the power converter were removed from its standard rectangular chassis and remounted on a small circular plate. The power converter is attached to the top of the battery package inside the instrumentation support tube. Figure 3-8 is an interconnection diagram of the transponder subsystem. Figure 3-9 is a block diagram


R3
2.2K
Q1
C1
C2
5-25
2N502
Y1
5000
575.000 KC
R1
47K
R2
12K
R4
4.7K
R5
47K
FROM RECEIVER
J4005
R3
2.2K
Q1
C1
C2
5-25
2N502
Y1
5000
570.000 KC
R1
47K
R2
12K
R4
4.7K
R5
47K

A



TELEMETRY OFF CHANNEL



TELEMETRY ON CHANNEL

B

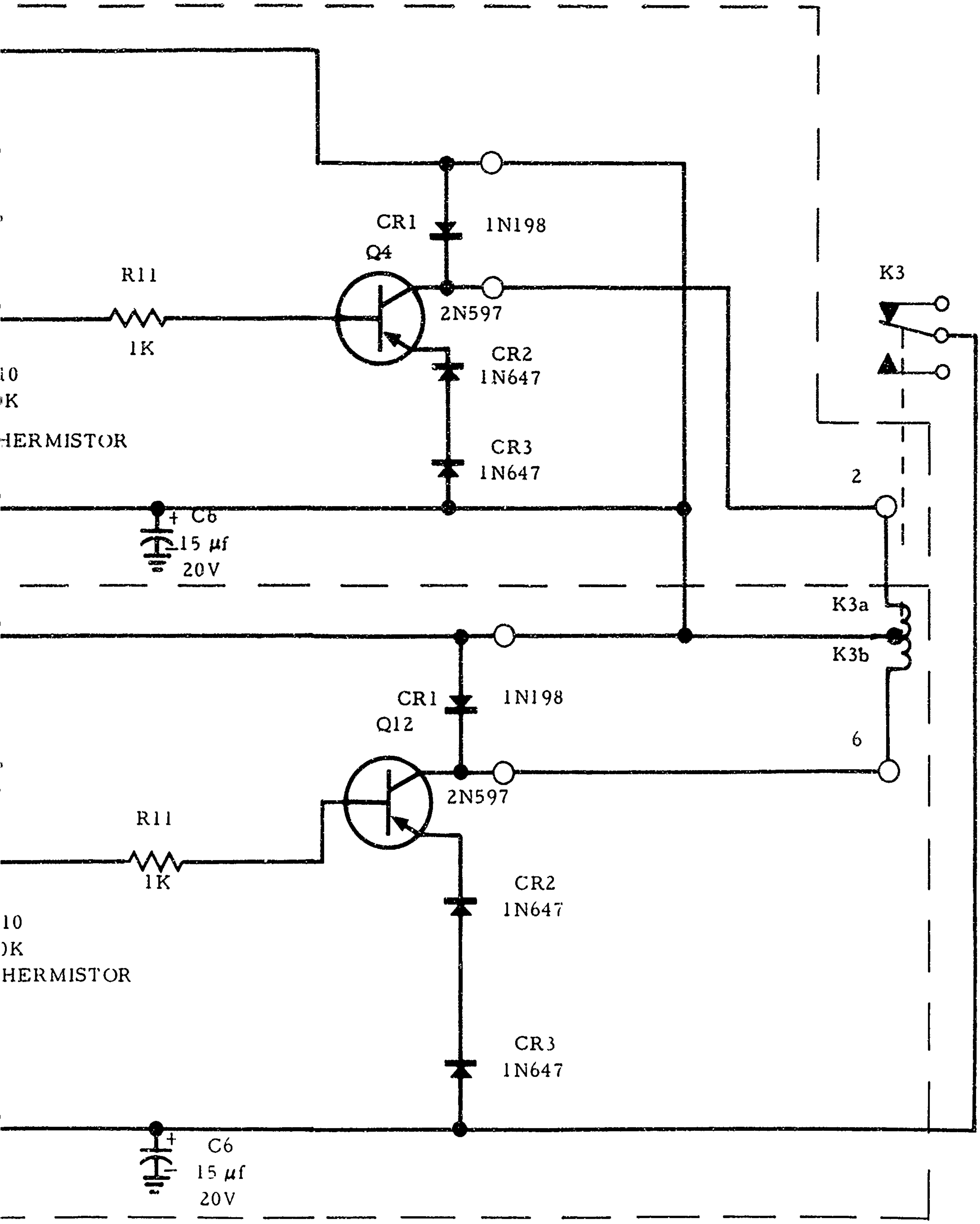


Figure 3-7. Schematic Diagram of Typical Telemetry Call Channel

C

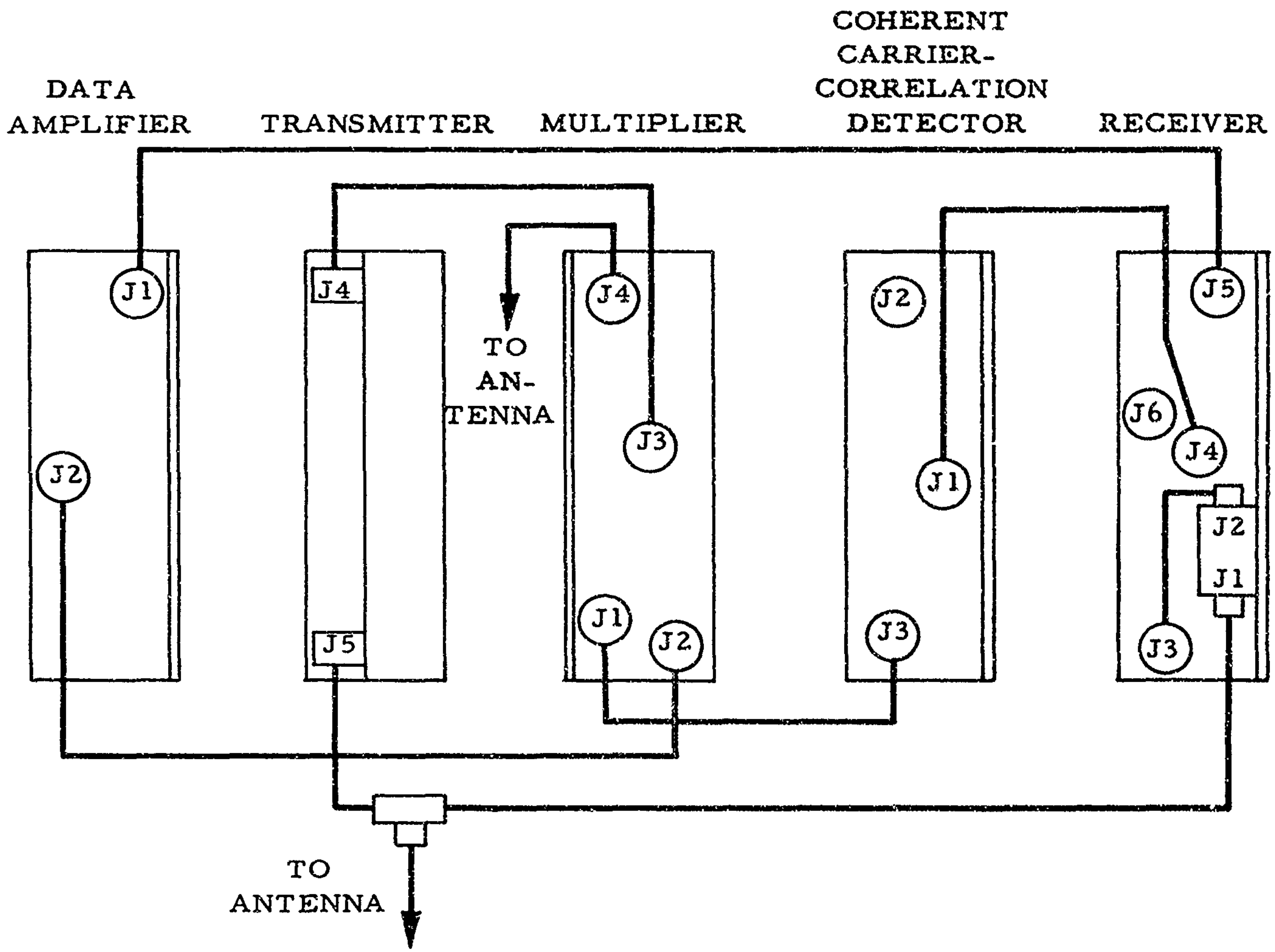


Figure 3-8. Transponder Subsystem Interconnections

of the transponder, and table II lists the major electrical and mechanical specifications.

3.5.4 Fabrication of Transponder Circuit Boards. Cubic fabricated the following circuit boards for each of the transponders and the SECOR ground station:

(1) 570 kc ~~mc~~ tone filter and amplifier

(2) 575 kc ~~mc~~ tone filter and amplifier

(3) 570 kc ~~mc~~ oscillator

(4) 575 kc ~~mc~~ oscillator.

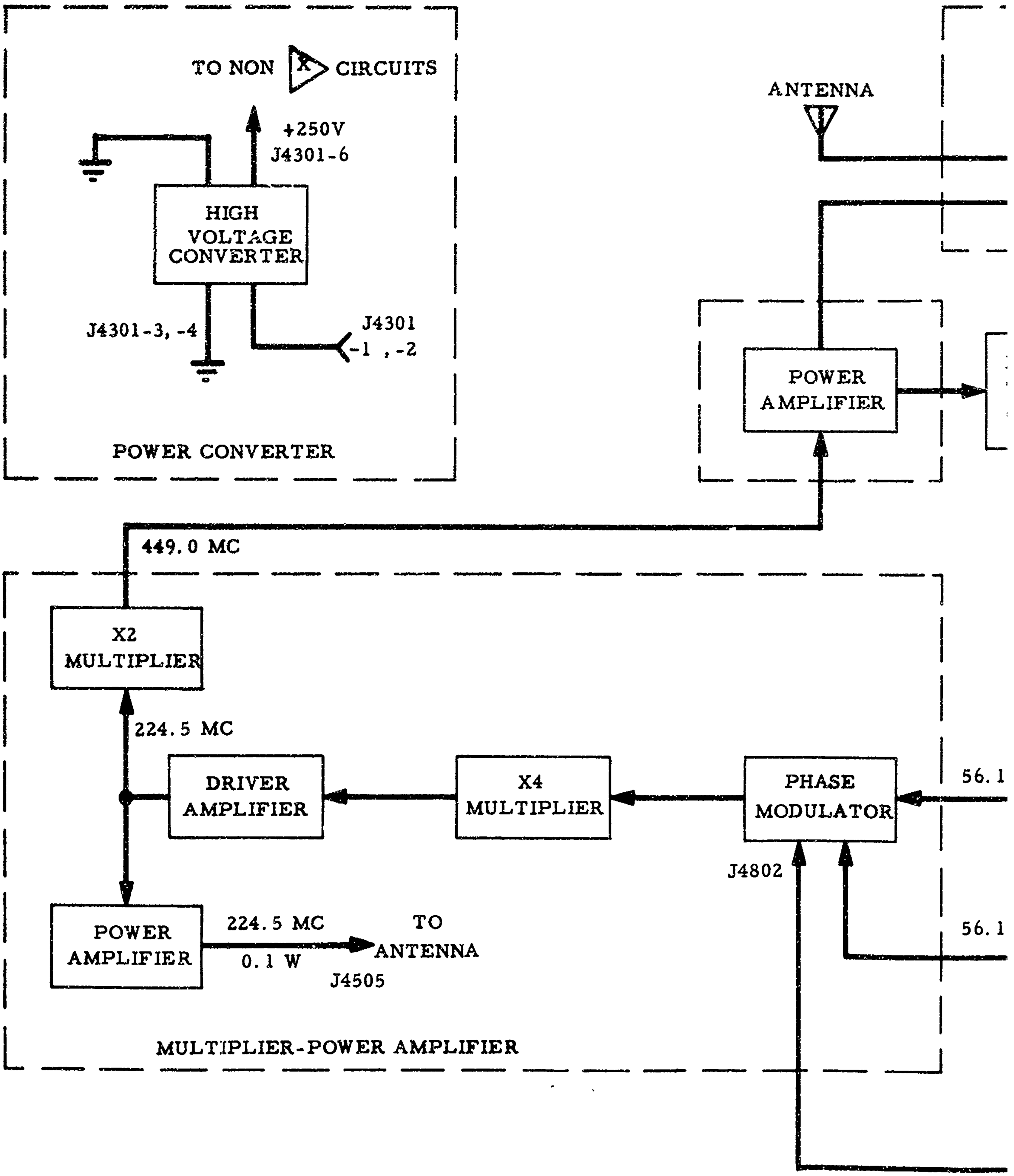
The amplifier boards are installed on the data amplifier and the receiver chassis. The oscillators are part of the geodetic SECOR ground stations. (See figure 3-10.)

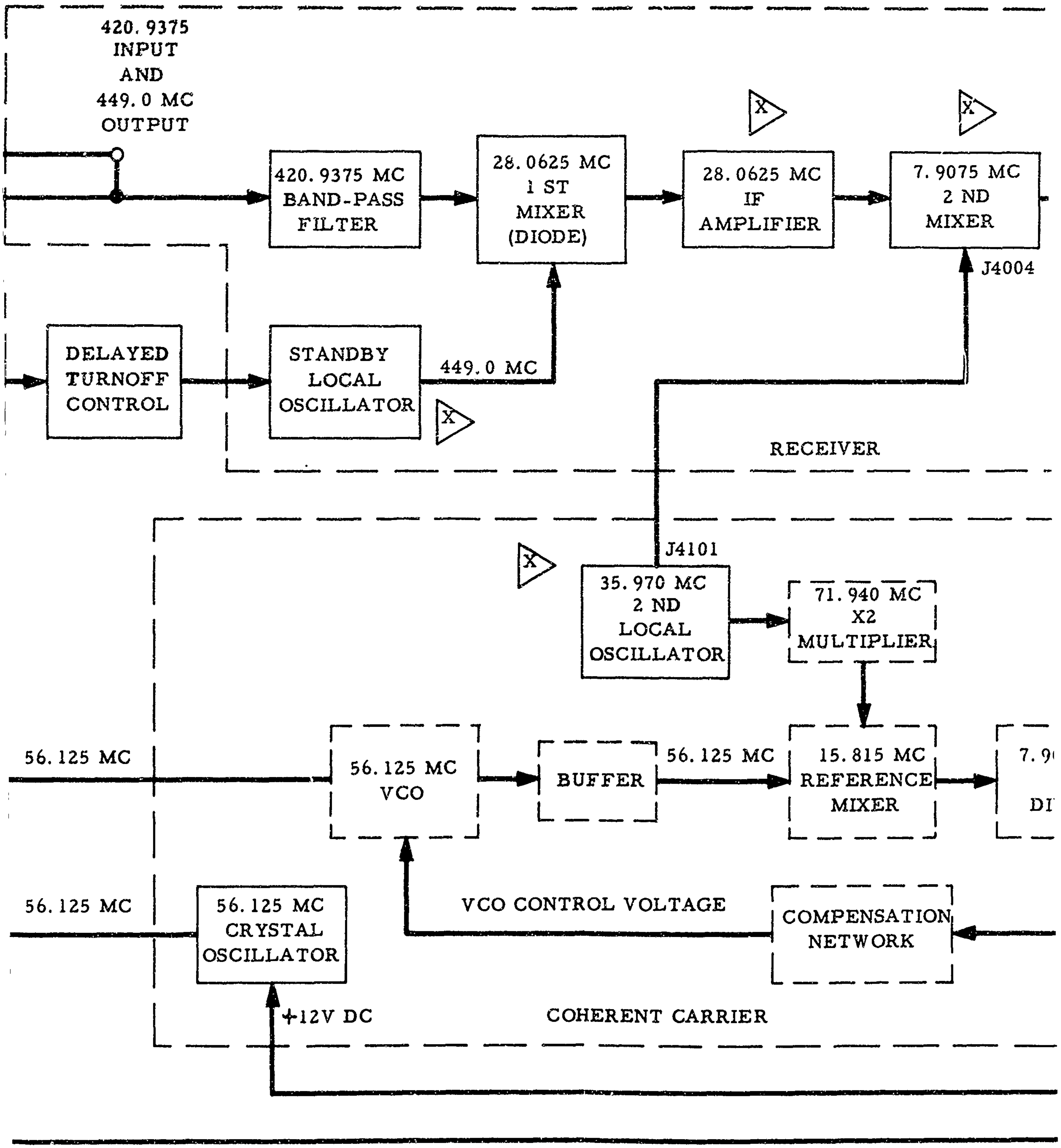
3.5.5 Transponder Subsystem Checkout. Cubic performed the following tests during the final assembly of the transponder subsystem. The tests are designed to insure the successful operation of the transponder subsystem and determine transponder suitability for operation in the spacecraft environment. They establish that the receiver has the proper sensitivity, that the transmitters have sufficient power output, and that the data signals are returned from the transponder in the correct manner. Table III is a list of the recommended test equipment; equivalent types may be substituted.

3.5.5.1 Transponder Calibration Test Setup. The transponder calibration test setup (figure 3-11) is as follows:

(1) Connect a directional coupler (Sierra 145 or equivalent) to a wattmeter (Bird 61 or equivalent).

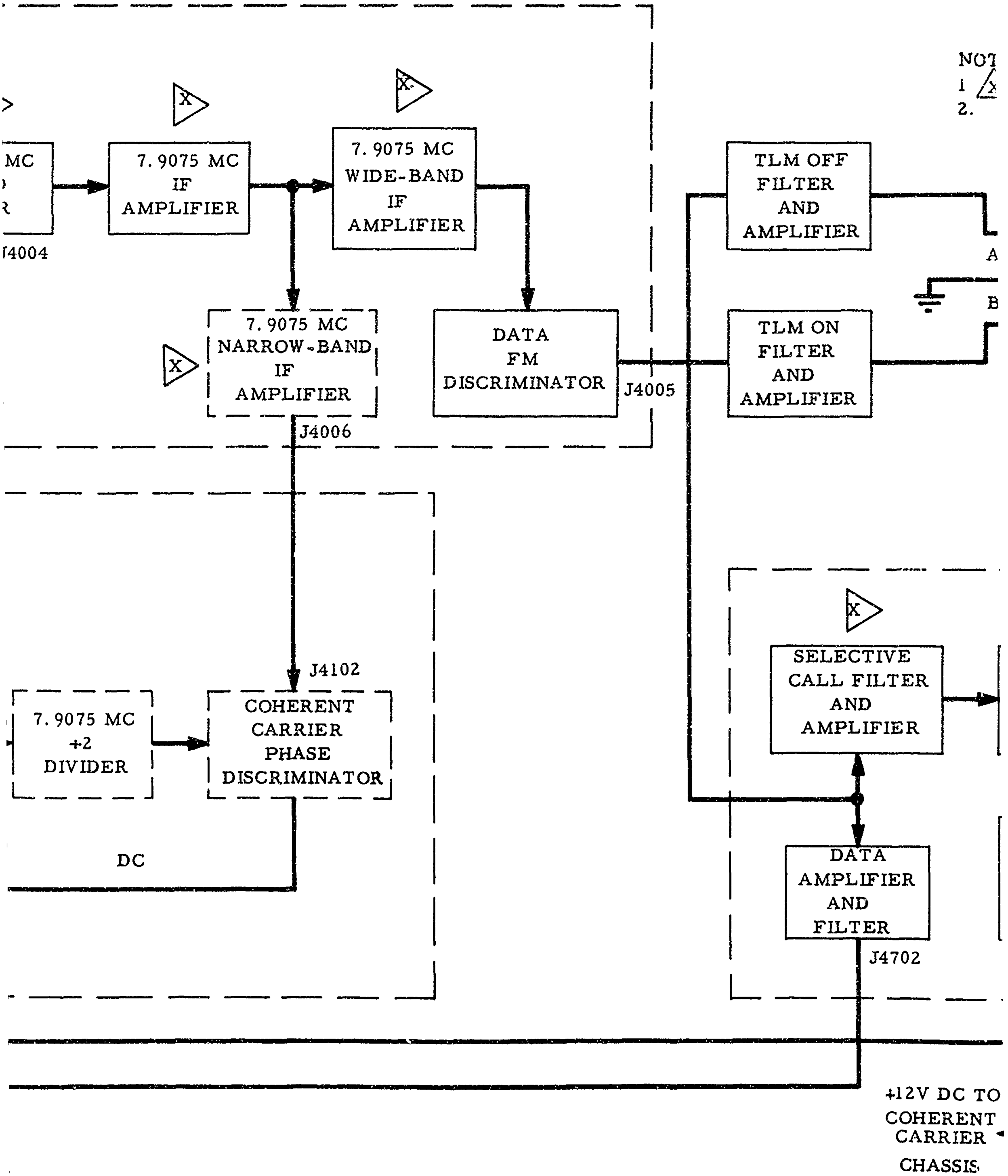
(2) Connect the SECOR test transmitter through a variable attenuator (Kay 30-0 or equivalent) to ER of the directional coupler.


TO NON
CIRCUITS
X
+250V
J4301-6
HIGH VOLTAGE CONVERTER
J4301-3, -4
J4301
-1 , -2
POWER CONVERTER
ANTENNA
POWER AMPLIFIER
449.0 MC
X2 MULTIPLIER
224.5 MC
DRIVER AMPLIFIER
X4 MULTIPLIER
PHASE MODULATOR
56.1
J4802
56.1
POWER AMPLIFIER
224.5 MC
0.1 W
TO ANTENNA
J4505
MULTIPLIER-POWER AMPLIFIER
A



420.9375
INPUT
AND
449.0 MC
OUTPUT
420.9375 MC BAND-PASS FILTER
28.0625 MC 1 ST MIXER (DIODE)
28.0625 MC IF AMPLIFIER
7.9075 MC 2 ND MIXER
J4004
DELAYED TURNOFF CONTROL
STANDBY LOCAL OSCILLATOR
449.0 MC
X
RECEIVER
J4101
35.970 MC 2 ND LOCAL OSCILLATOR
71.940 MC X2 MULTIPLIER
56.125 MC
56.125 MC VCO
BUFFER
56.125 MC
15.815 MC REFERENCE MIXER
56.125 MC
56.125 MC CRYSTAL OSCILLATOR
VCO CONTROL VOLTAGE
COMPENSATION NETWORK
+12V DC
COHERENT CARRIER

13



MC
J4004
7.9075 MC
IF
AMPLIFIER
7.9075 MC
WIDE-BAND
IF
AMPLIFIER
7.9075 MC
NARROW-BAND
IF
AMPLIFIER
J4006
DATA
FM
DISCRIMINATOR
J4005
TLM OFF
FILTER
AND
AMPLIFIER
TLM ON
FILTER
AND
AMPLIFIER
A
B
J4102
7.9075 MC
÷2
DIVIDER
COHERENT
CARRIER
PHASE
DISCRIMINATOR
DC
SELECTIVE
CALL FILTER
AND
AMPLIFIER
DATA
AMPLIFIER
AND
FILTER
J4702
+12V DC TO
COHERENT
CARRIER
CHASSIS

NOTES:
1. [X] STAGES ENERGIZED IN STANDBY OPERATION
2. STAGES UTILIZED IN THE TR-17 TRANSPONDER ONLY ARE REPRESENTED BY DASHED LINES.

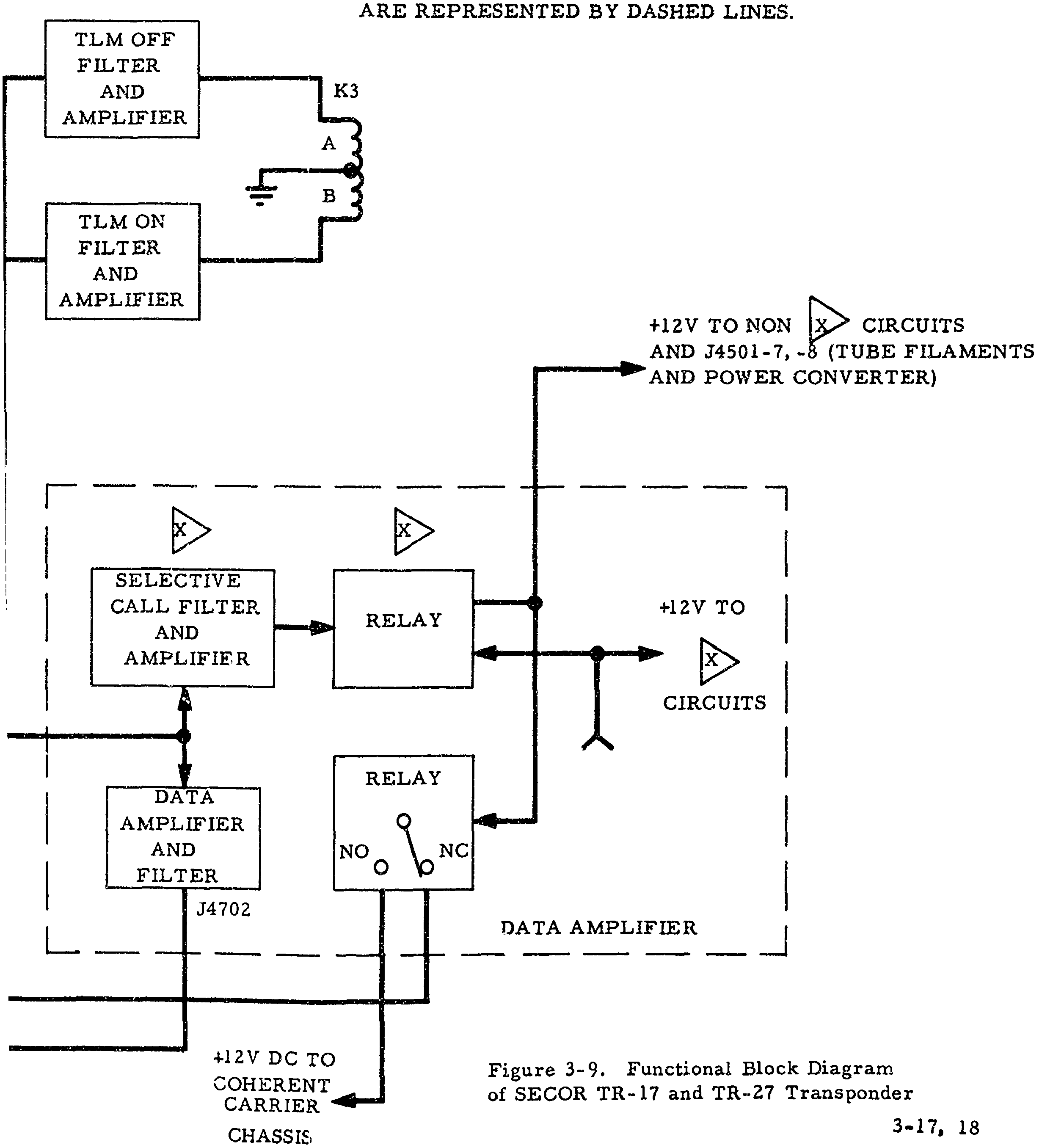


Figure 3-9. Functional Block Diagram of SECOR TR-17 and TR-27 Transponder

D

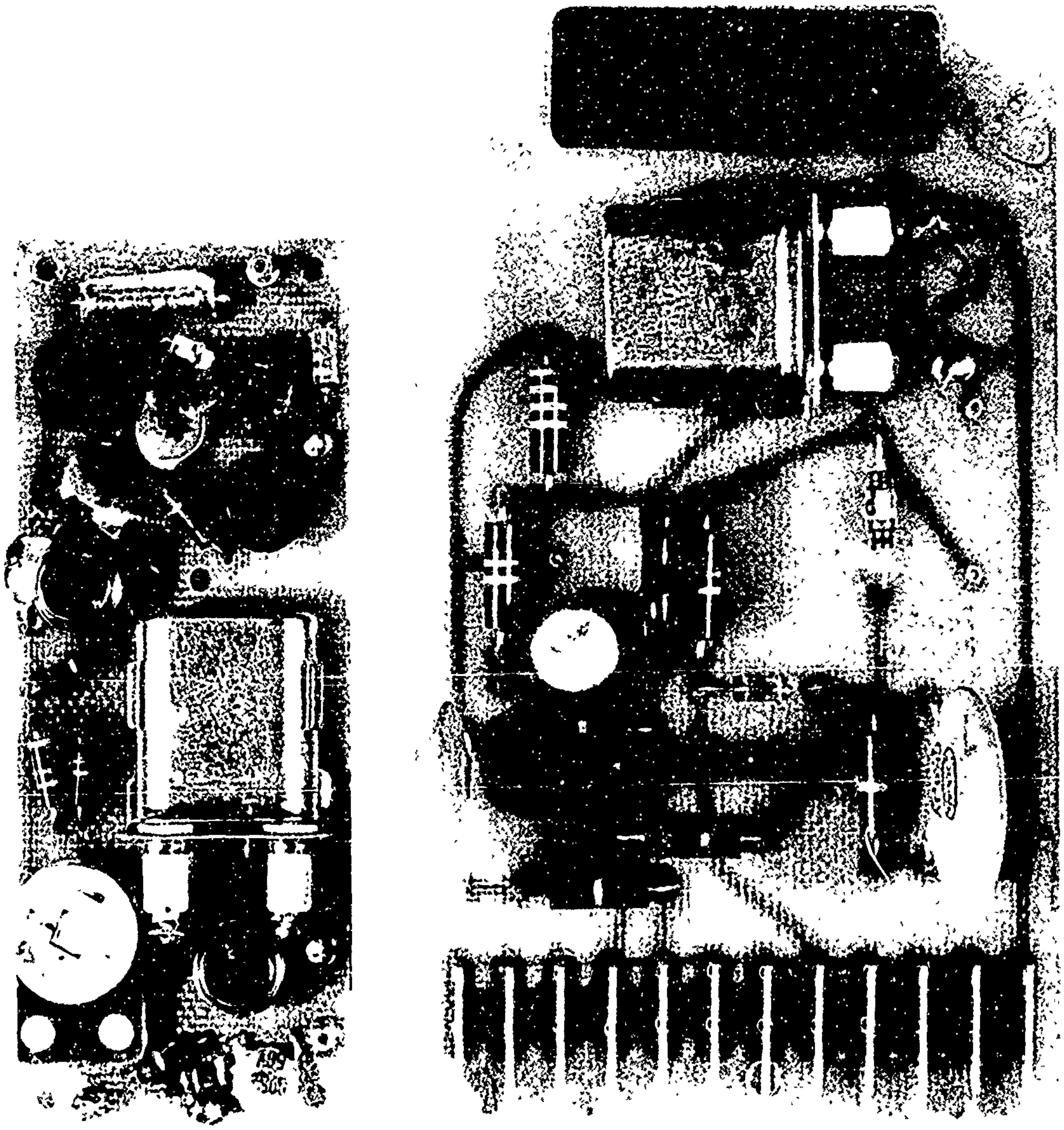
TABLE II

PHYSICAL AND ELECTRICAL SPECIFICATIONS
FOR
TRANSPONDER SUBSYSTEM

Frequencies	Interrogate 420.9375 mc ±(10 pps plus doppler)
	Respond 449.0 mc ±(50 ppm plus doppler)
	Respond 224.5 mc ±(50 ppm plus doppler)
Transmitter bandwidth	1.5 mc
Receiver sensitivity	106 dbm
Power output	1 watt minimum at 449.0 mc into antenna
	1 watt minimum at 224.5 mc into antenna
Antenna Required	50 ohms
	VSWR 1.5
Maximum allowable phase shift through transponder	±1 degree
Chassis weights	
Coherent carrier - correlation detector	0.582 pound
Receiver	0.668
Data amplifier	0.618
Multiplier	0.485

TABLE II (Cont)

Chassis weights (Cont)			
Transmitter	0.770		
Power Converter	0.731		
	3.85 pounds		
Power Dissipation (watts)			
	Standby	Energized	Selectively called
Transistor units	0.2	0.35	0.8
Vacuum tube units	0	0	20.7
Power converter unit	0	0	5.5
Total	0.2	0.35	27.0



TONE FILTER AND AMPLIFIER TONE OSCILLATOR

Figure 3-10. Typical Telemetry Call Printed Circuit Boards

TABLE III

TEST EQUIPMENT UTILIZED FOR TRANSPONDER CHECKOUT

Description	Manufacturer	Model No.
SECOR test transmitter	Cubic	TS5
Variable attenuator	Kay	30-0
Directional coupler	Sierra	145
Wattmeter	Bird	61
Frequency counter	Beckman-Berkely	7370
Amplifier		7570
Converter		7573
Converter		7572
Converter		7571
Milliwattmeter	Hewlett-Packard	430C
RF generator	Hewlett-Packard	606A
Spectrum analyzer	Polarad	DU
		(with STU-1A plug-in unit)
RF generator	Hewlett-Packard	608C

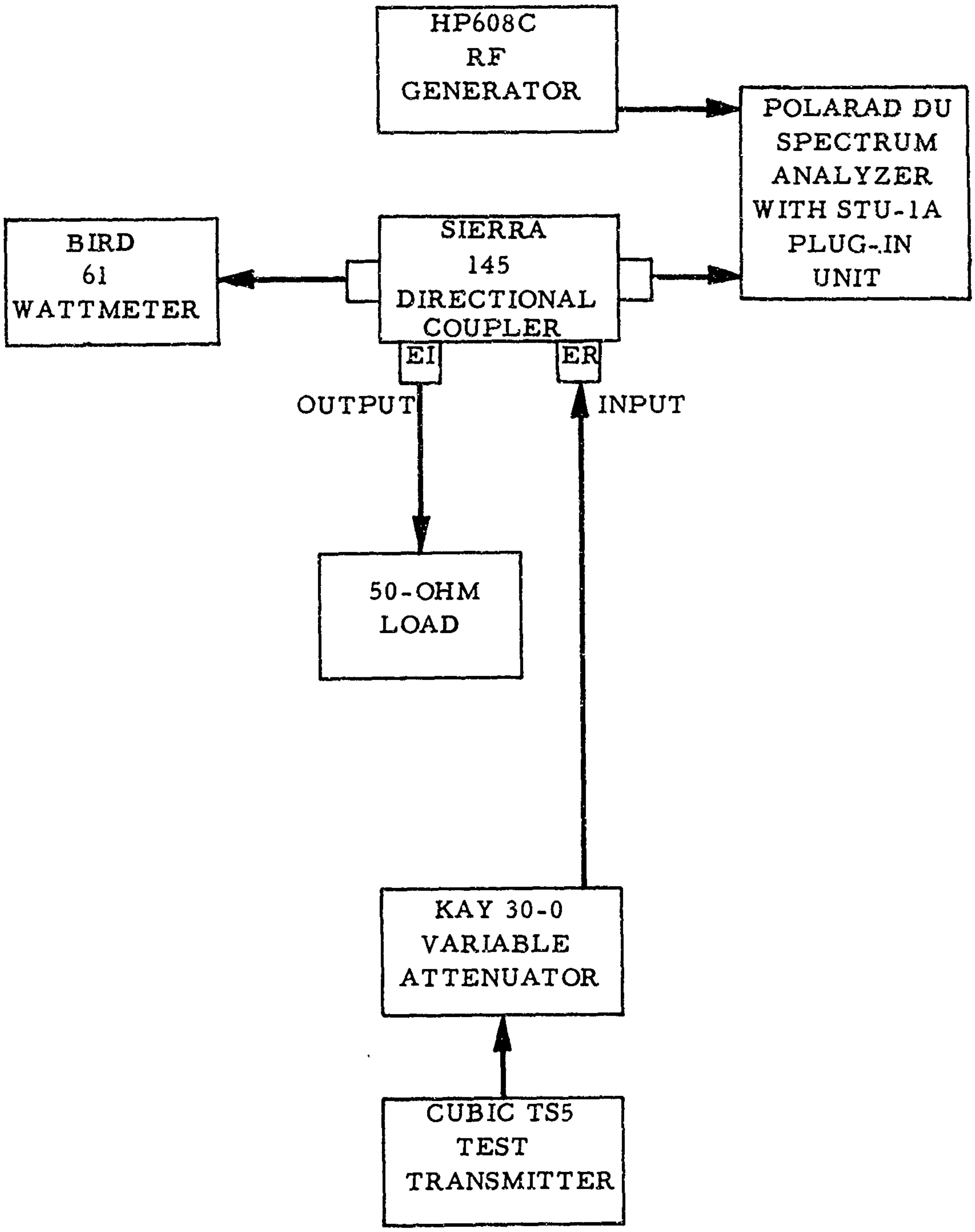


Figure 3-11. Transponder Calibration Test Setup

(3) Connect the input of the directional coupler to a spectrum analyzer (Polarad DU with STU-1A plug-in unit or equivalent).

(4) Adjust the gain of the spectrum analyzer and the variable attenuator for a suitable display on the analyzer.

(5) Note the amplitude of this display.

(6) Disconnect the spectrum analyzer from the directional coupler and connect the signal generator (HP 608C or equivalent) to the spectrum analyzer.

(7) Adjust the signal generator to 420.9375 mc and adjust the output for the same deflection as noted in step (5) above.

(8) The output reading of the signal generator is the signal level reaching the transponder from the SECOR test transmitter. This may be readjusted to a higher or lower level by tuning the variable attenuator.

3.5.5.2 Adjustment of Test Transmitter Modulation Index. The test transmitter modulation index adjustment test setup (figure 3-12) is as follows:

(1) Perform steps (1) through (5) outlined in paragraph 3.5.5.1

(2) Connect an rf generator (HP 606A or equivalent) to the modulation input of the SECOR test transmitter.

(3) Adjust the rf generator to approximately 500 kc.

(4) Adjust the level of the rf generator until the first order sidebands are 0.2 of the carrier level. This corresponds to modulation index of 0.4.

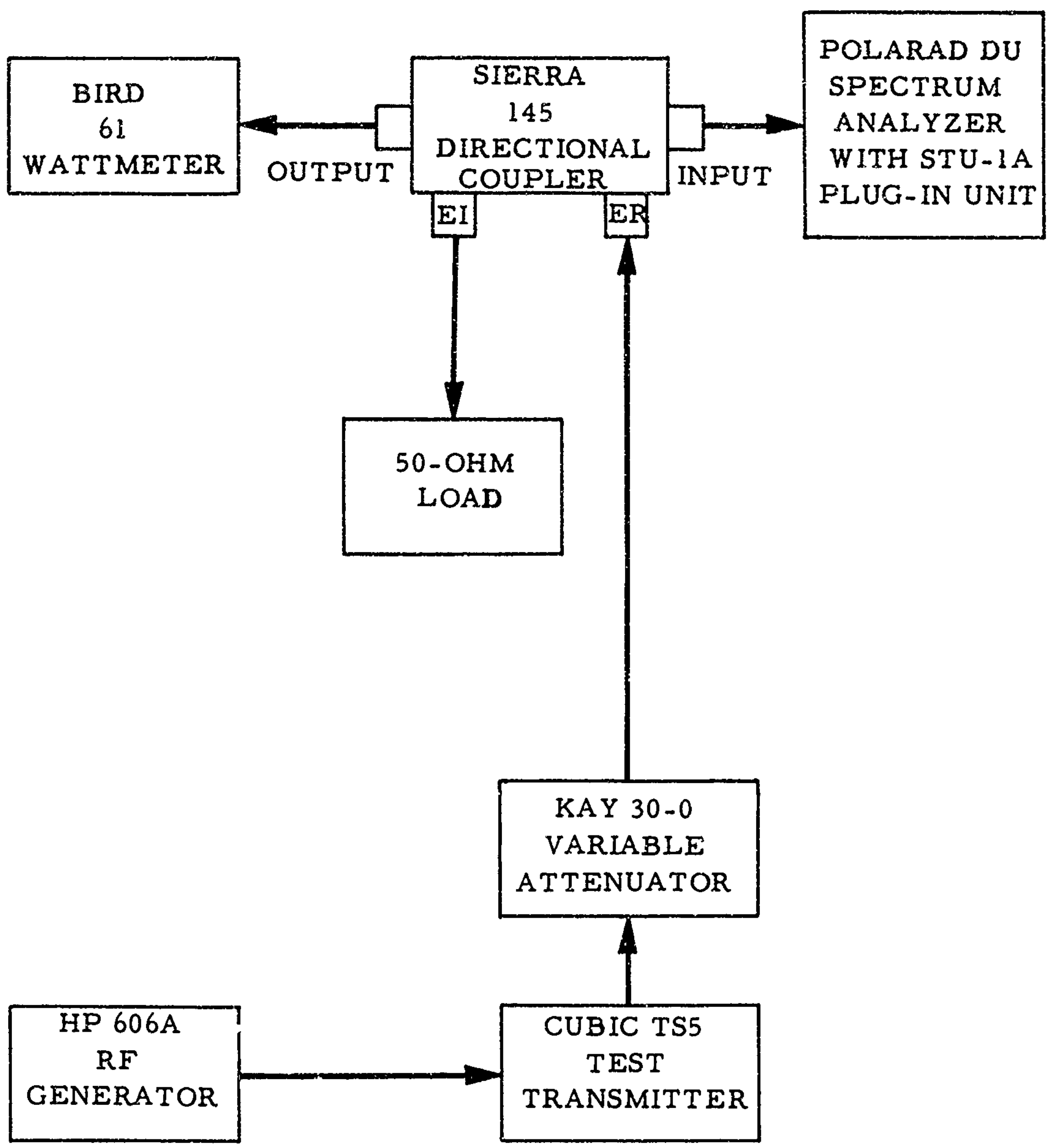


Figure 3-12. Test Setup for Adjustment of Test Transmitter Modulation Index

3.5.5.3 Transponder Checkout. The test setup for the checkout of the transponder is shown in figure 3-13. The checkout procedure is as follows:

(1) Connect the transponder HF output to the input of a directional coupler (Sierra 145 or equivalent) and to the wattmeter (Bird 61 or equivalent); connect the transponder LF output to a milliwattmeter (HP 430C or equivalent).

(2) Connect a spectrum analyzer (Polarad DU with STU-1A plug-in unit, or equivalent) to EI of the directional coupler.

(3) Connect the SECOR test transmitter through a variable attenuator (Kay 30-0 or equivalent) to ER of the directional coupler.

(4) Connect an rf generator (HP 606A or equivalent) to the modulation input of the test transmitter.

(5) Adjust for an index of 0.4. (Refer to step (4) of paragraph 3.5.5.2.)

(6) Turn on the transponder.

(7) Turn on the test transmitter.

(8) Turn on the select-call modulation of the test transmitter.

(9) Allow two minutes warm up.

(10) The HF power output shall be 1.0 watt minimum.

(11) The LF power output shall be 100 mw minimum.

(12) Adjust the variable attenuator for -95 dbm signal input to the transponder.

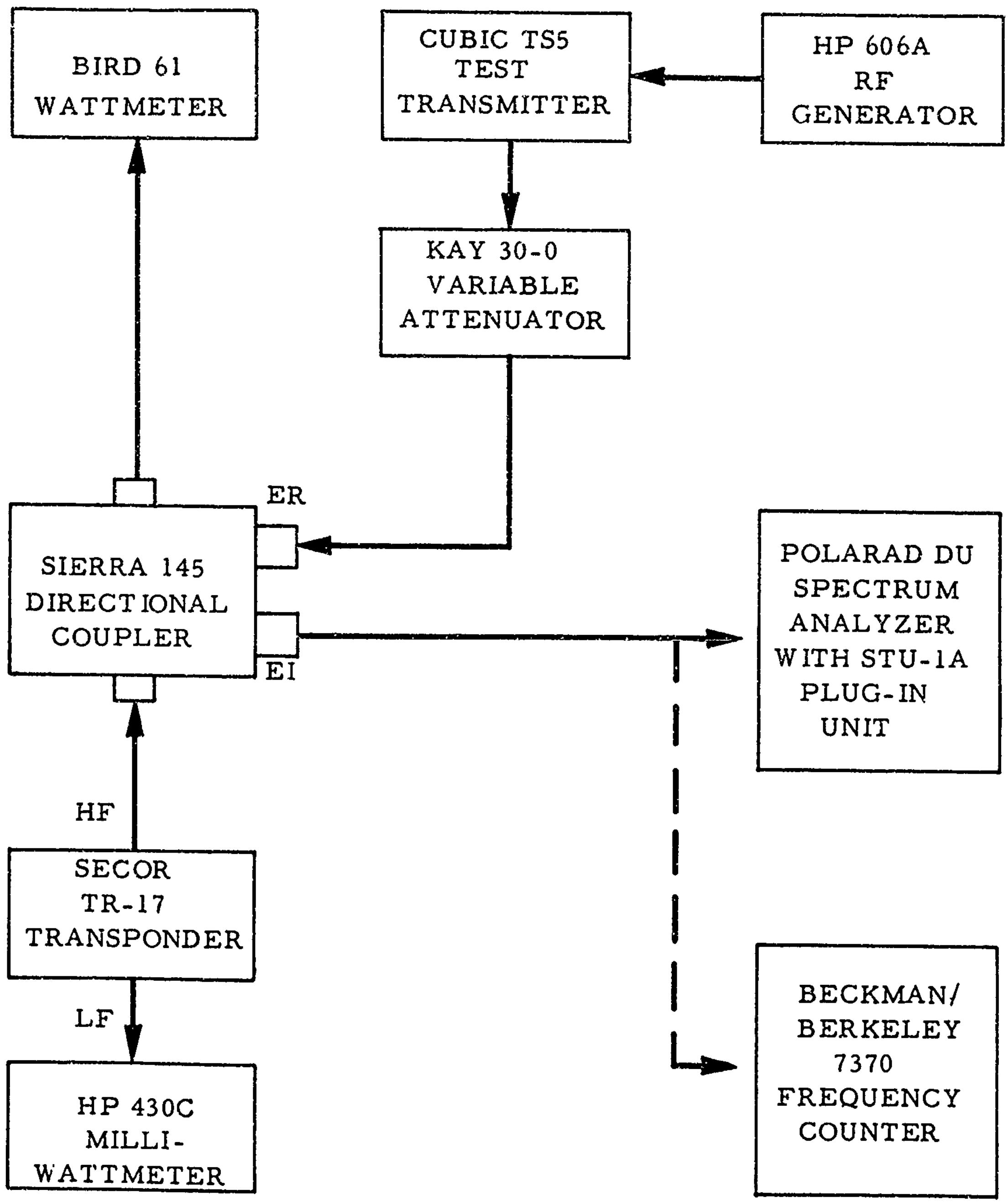


Figure 3-13. Test Setup for Transponder Checkout

(13) Check the select-call operation by removing and reapplying the select-call modulation.

(14) Check the data return on the spectrum analyzer (tuned to 449.0 mc) by tuning the HP 606A rf generator across each of the following data frequencies:

585.533 mc.
585.246 mc.
565.000 mc.
549.223 mc.
548.937 mc.

(15) Note the returned data sidebands on the spectrum analyzer.

(16) Disconnect the spectrum analyzer from the directional coupler and connect the analyzer to the LF output of the transponder.

(17) Repeat step (14) above, but tune the spectrum analyzer to 224.5 mc.

(18) Disconnect the test transmitter.

(19) Connect a frequency counter (Beckman/Berkeley 7370 with appropriate auxiliary units, or equivalent) to EI of the directional coupler.

(20) Determine the frequency of the HF signal.

(21) The HF frequency shall be 449.000 mc ±0.0025 per cent.

(22) Remove the power and disconnect the test equipment.

3.6 Developmental History of the Telemetry Subsystem. Cubic designed, fabricated and installed a telemetry subsystem in each of the Geodetic Spacecraft. The calibration and tests were performed on each functional chassis to insure that the reliability and specifications were satisfactory for this intended use.

3.6.1 General Problem. The Geodetic Spacecraft required a small transistorized telemeter operating in accordance with NASA specifications to monitor spacecraft shell and internal temperatures, battery compartment temperature, and battery voltage, as well as specific data concerning the transponder operation. At a meeting with NASA on 8 March 1961, IRIG subcarrier channel No. 3 was selected for the Geodetic Spacecraft. The U. S. Army Radio Frequency Coordination Group assigned 136.11 mc as the transmission fr.quency.

3.6.2 Preliminary Considerations. Upon analysis of the problem, it was apparent that a solid-state FM/PM telemetry subsystem using a commutator similar to the NRL Lofti design would best fulfill the spacecraft requirements. The telemetry subsystem consists of the individual temperature and voltage sensing elements, an eight-channel commutator, a voltage-controlled oscillator (VCO), and a transmitter. It was also necessary to consider the available space. It was decided to design the subsystem to fit on five printed circuit boards (2-7/8 inches x 4-5/8 inches) which would attach to the support tube above the transponder chassis.

3.6.3 Development Details. The following paragraphs describe the work accomplished in the development of each of the functional telemeter modules. Figure 3-14 is a functional block diagram of the subsystem.

3.6.3.1 Development of the Sensing Circuits. The telemetry subsystem required sensing elements for each of the following parameters:

(1) 2V calibration developed from +3V internal supply

(2) 4V calibration developed from +6V internal supply

(3) Battery voltage of $12.0^{+2.0}_{-1.0}$V

(4) SECOR transponder plate current with $7.67^{+1.11}_{-1.40}$V measured across the cathode resistor

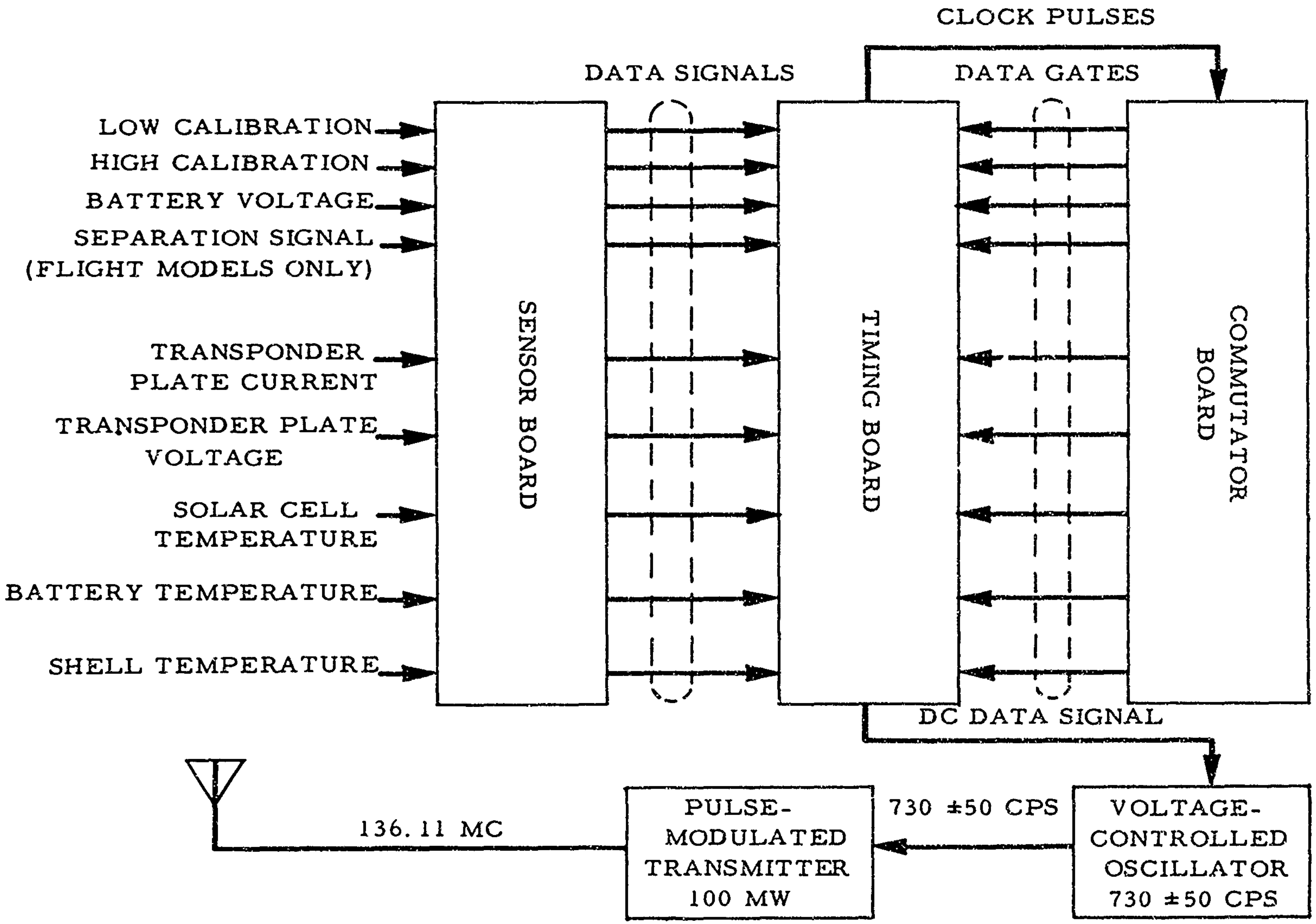


Figure 3-14. Functional Block Diagram of the Telemetry Subsystem

(5) SECOR transponder plate voltage of $250\,^{+20}_{-10}\,V$

(6) Solar cell temperature

(7) Battery temperature

(8) Shell temperature

(9) Non-separation signal (flight models only).

Specifications for the various elements are as follows:

(1) Operating range - Temperature, $0^{\circ}C$ to $50^{\circ}C$
- Battery voltage, 9V to 15V

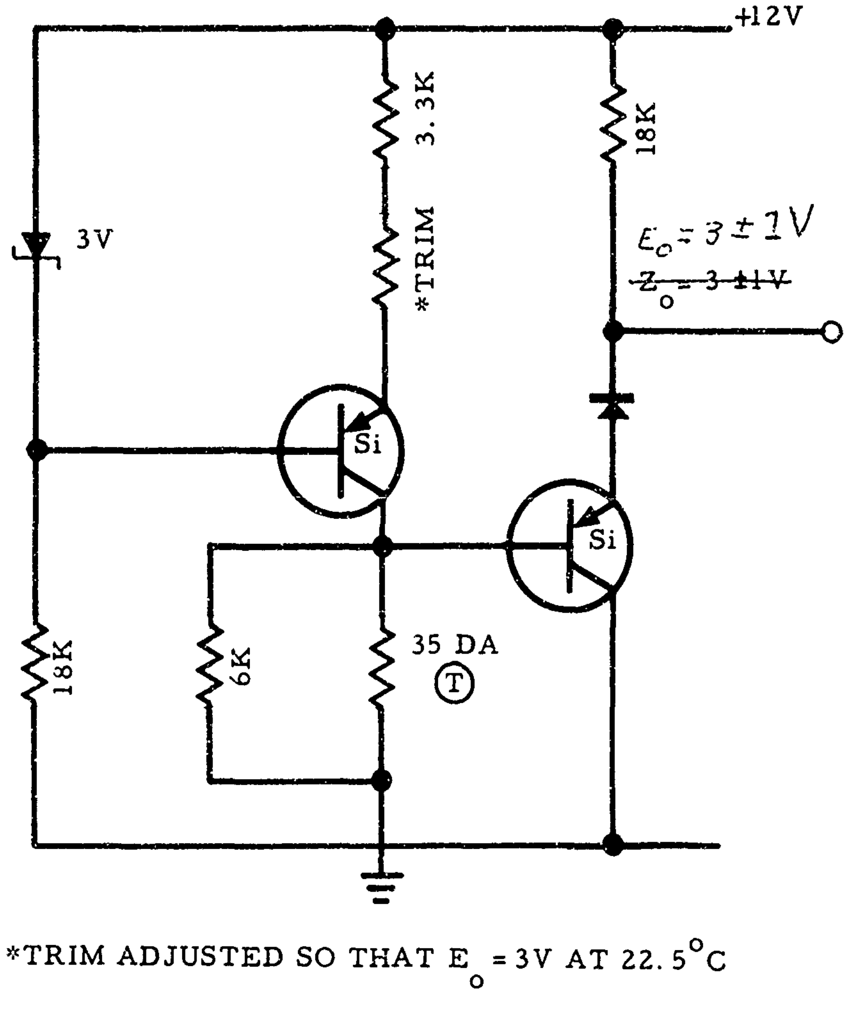
(2) Accuracy - 5 per cent

(3) Output range - 3 ±1V dc

(4) Output impedance - less than 1K

Cubic selected Veco 35D4 thermistors for the temperature-sensing elements, and one per cent sensing resistors for the voltage and current measurements. A special circuit including microswitches is required to provide a spacecraft non-separation signal. This signal will appear as a spike on the battery voltage measurement.

3.6.3.1.1 The characteristics of the sensing circuits were primarily determined by the allowable current drain and the voltages to be measured. The thermistor temperature-sensing circuits were designed to provide a change of 2V over the temperature range with minimum nonlinearity. This problem was solved by supplying the thermistors from constant current generators. The potential across the thermistor then is directly proportional to the thermistor resistance. This technique avoided the need for elaborate compensation networks to obtain the linear functions. The thermistor potential is detected by an emitter follower which provides the required low output impedance to the commutator. Figure 3-15 shows a typical temperature-sensing circuit.



*TRIM ADJUSTED SO THAT $E_o = 3V$ AT 22.5°C

Figure 3-15. Typical Temperature-Sensing Circuit

3.6.3.1.2 The individual thermistors are attached to the shell, a solar cell, and the battery package. The rest of the temperature-sensing circuitry is placed on a separate sensor circuit board. (See figure 3-16.) On this board also are the compensating and impedance-matching components for the voltage and current measuring elements. Figure 3-17 is a schematic diagram of the sensor board. The non-separation signal is provided by a commutator clock pulse which is connected to channel No. 3 (battery voltage) and powered through the microswitch ~~through two microswitches.~~ The switches, located on the base of the internal thrust structure, are held closed by spring-loaded pins which extend through the separation fixture and rest on the vehicle separation plate. When separation occurs, the pins extend, opening the switches, and the spike is removed from channel No. 3. Figur 3-18 illustrates the proposed separation signal fixture.

3.6.3.2 Development of the Commutator. The commutator circuits are based on a design developed by NRL for the Lofti spacecraft. The Cubic design consists of a 1 pps clock, a three-stage binary counter or switch, a +3V and a +6V voltage regulator and a multiplexing network. The solid-state commutator samples each of the eight parameters once each 6.6 seconds and presents the individual dc voltage variations to the VCO. The commutator specifications are:

(1) Number of channels: 8

(2) Frame synchronization: double pulse, high-low calibration

(3) Sampling rate: 1 sample per second ± 0.2 sec

(4) Sampling duration: 1 ±0.2 sec

(5) Signal level: +3 ±1V dc

(6) Source impedance: 0 to 10,000 ohms

(7) Reverse leakage: back current through sampled source less than 1 microampere

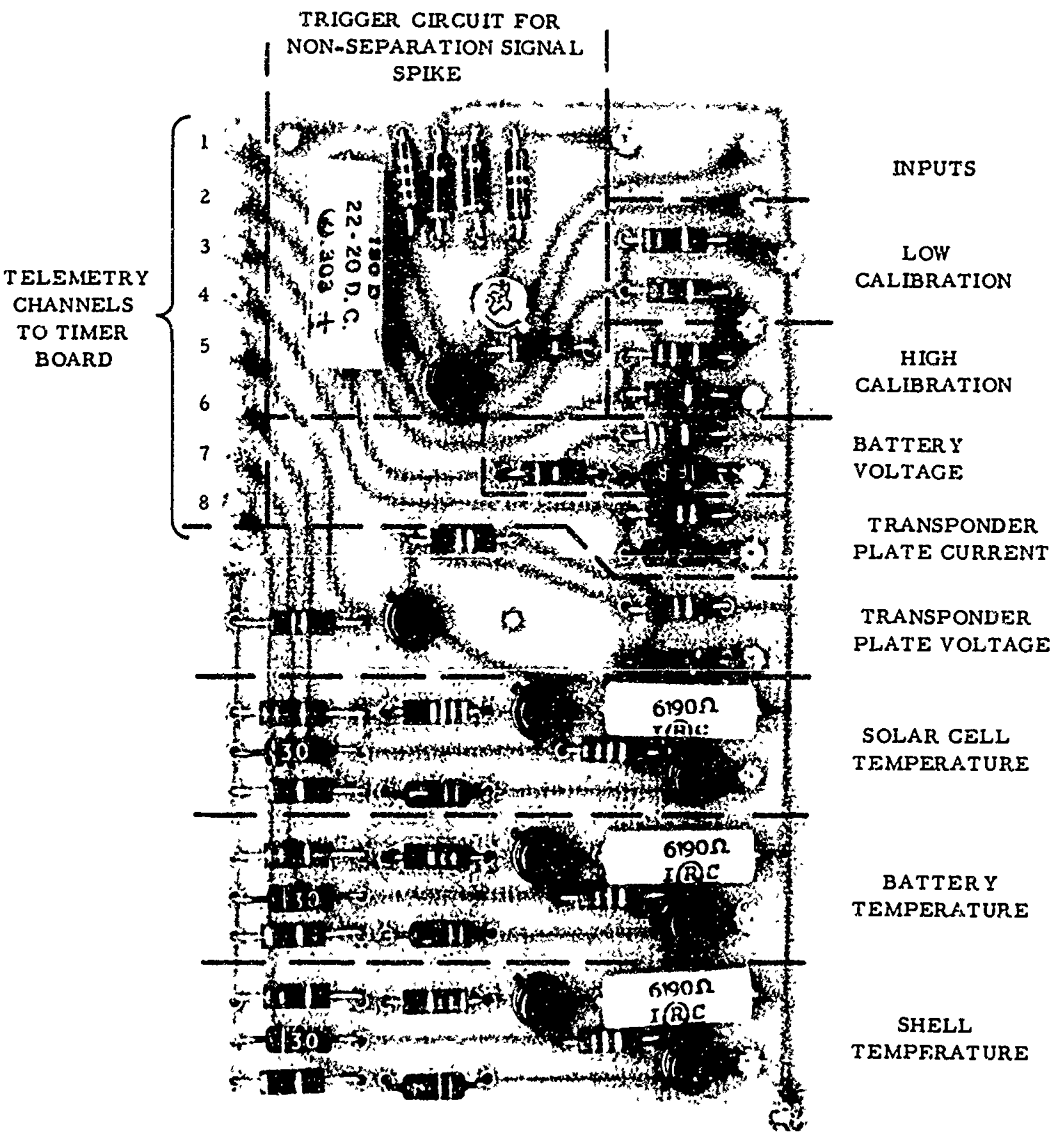


Figure 3-16. Typical Sensor Printed Circuit Board

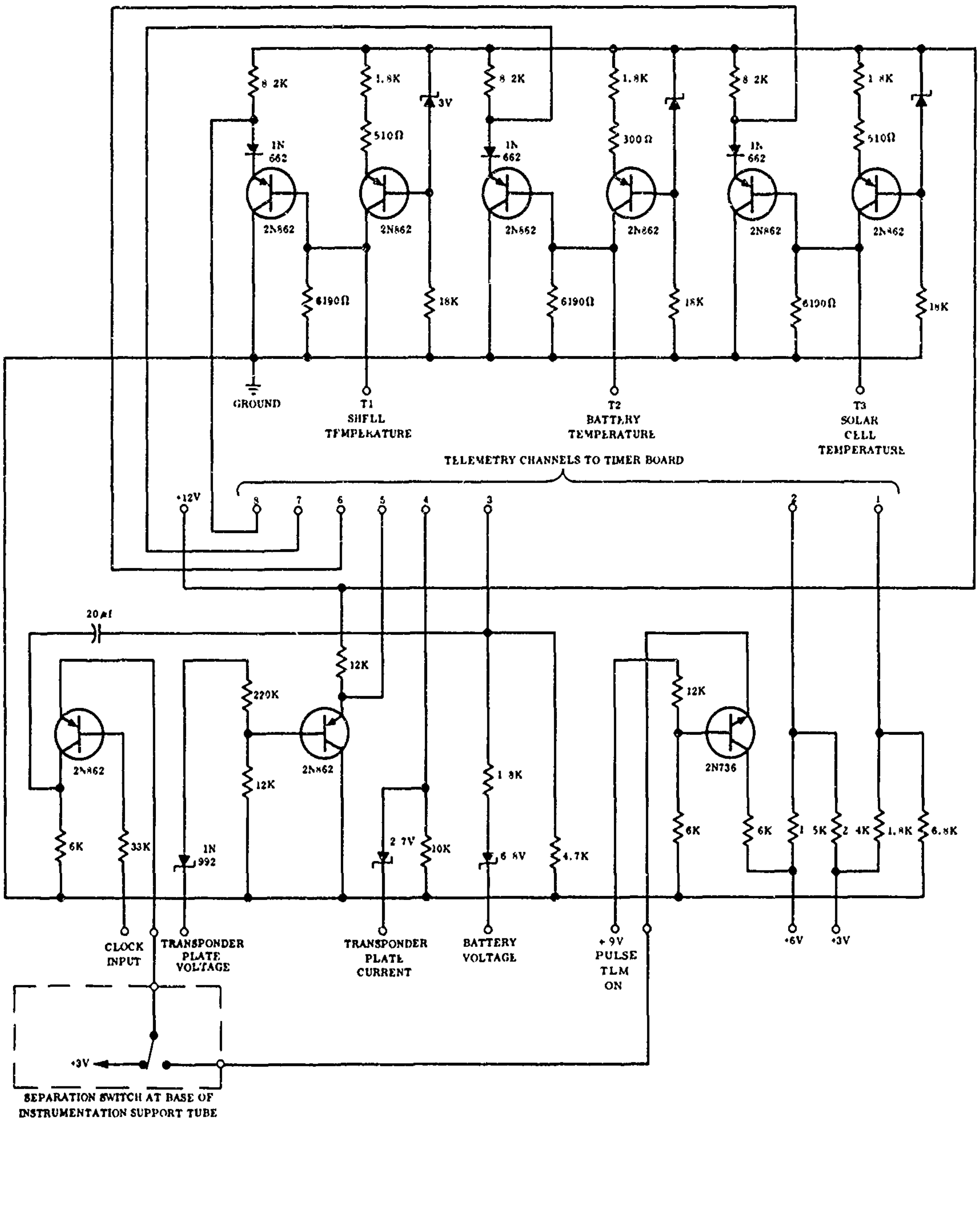


Figure 3-17. Schematic Diagram of the Sensor Board

(8) Cross talk: 0.1 per cent of full scale

(9) Independent linearity error: 0.3 per cent

(10) Stability: ±0.01 per cent for four hours at constant temperature

(11) Channel uniformity: +0.03 per cent after setting time

(12) Output range: +3 ±1V dc

(13) Output impedance: source impedance (plus approximately 600 ohms) in parallel with 300K

(14) Power requirements: +12V ±10 per cent dc.

The commutator was placed on a separate circuit board (figure 3-19) to maintain a low component density. Figures 3-20 and 3-21 are schematic diagrams of the commutator and the timer, resepectively.

3.6.3.3 Development of the Voltage-Controlled Oscillator (VCO). The VCO produces a change in its output frequency proportional to the change in the input voltage. The unit operates from the output of the commutator and provides signals to the modulation section of the transmitter. The VCO specifications are as follows:

(1) Input: 3V ±1V dc

(2) Sensing: frequency increases as the input voltage changes in a pos:.ive direction

(3) Input impedance: 0.1 (1 - j6) megohms

(4) Input intelligence frequency response: 1 per cent

(5) Output amplitude: 3V rms open circuit

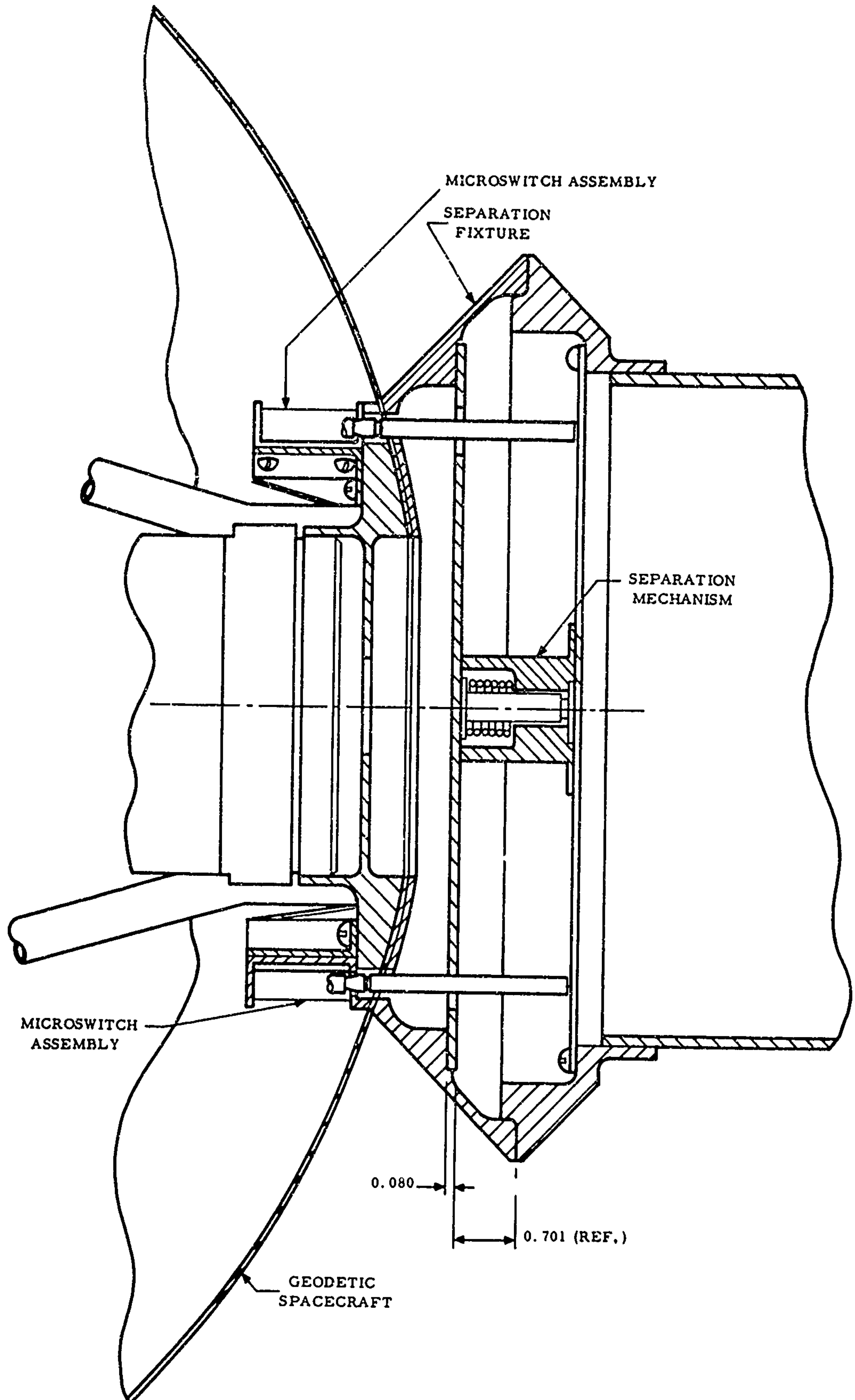


Figure 3-18. Separation Signal Fixture

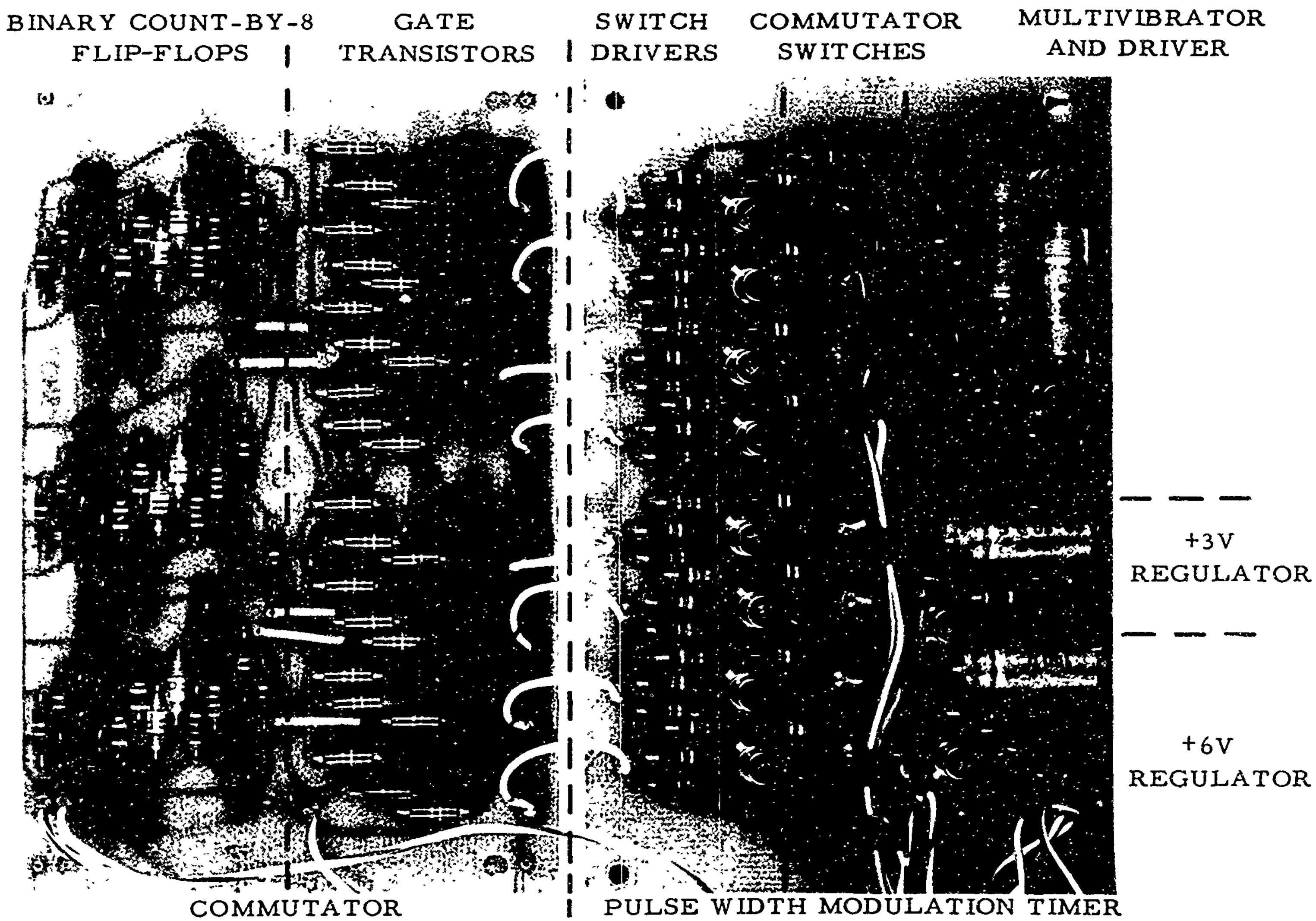


Figure 3-19. Typical Commutator and Timer Printed Circuit Boards

(6) Output impedance: 100K isolation impedance at channel frequency

(7) Output amplitude modulation: amplitude modulation of the oscillator output signal, with frequency deviation over the bandwidth does not exceed 1 db

(8) Output distortion: the total harmonic distortion content of the output waveform is less than 1.0 per cent of the maximum output voltage.

(9) Linearity: output frequency versus applied input signal does not deviate more than ±0.25 per cent of the bandwidth from the best straight line.

(10) Stability: under static environmental condition; ± 0.25 per cent of the bandwidth, 15 minutes to 8 hours at +25°C

(11) Power requirements: +12V dc ±10 per cent

(12) Supply voltage variation: subcarrier frequency and sensitivity remain constant to 1.0 per cent with supply variation of 1.0 per cent.

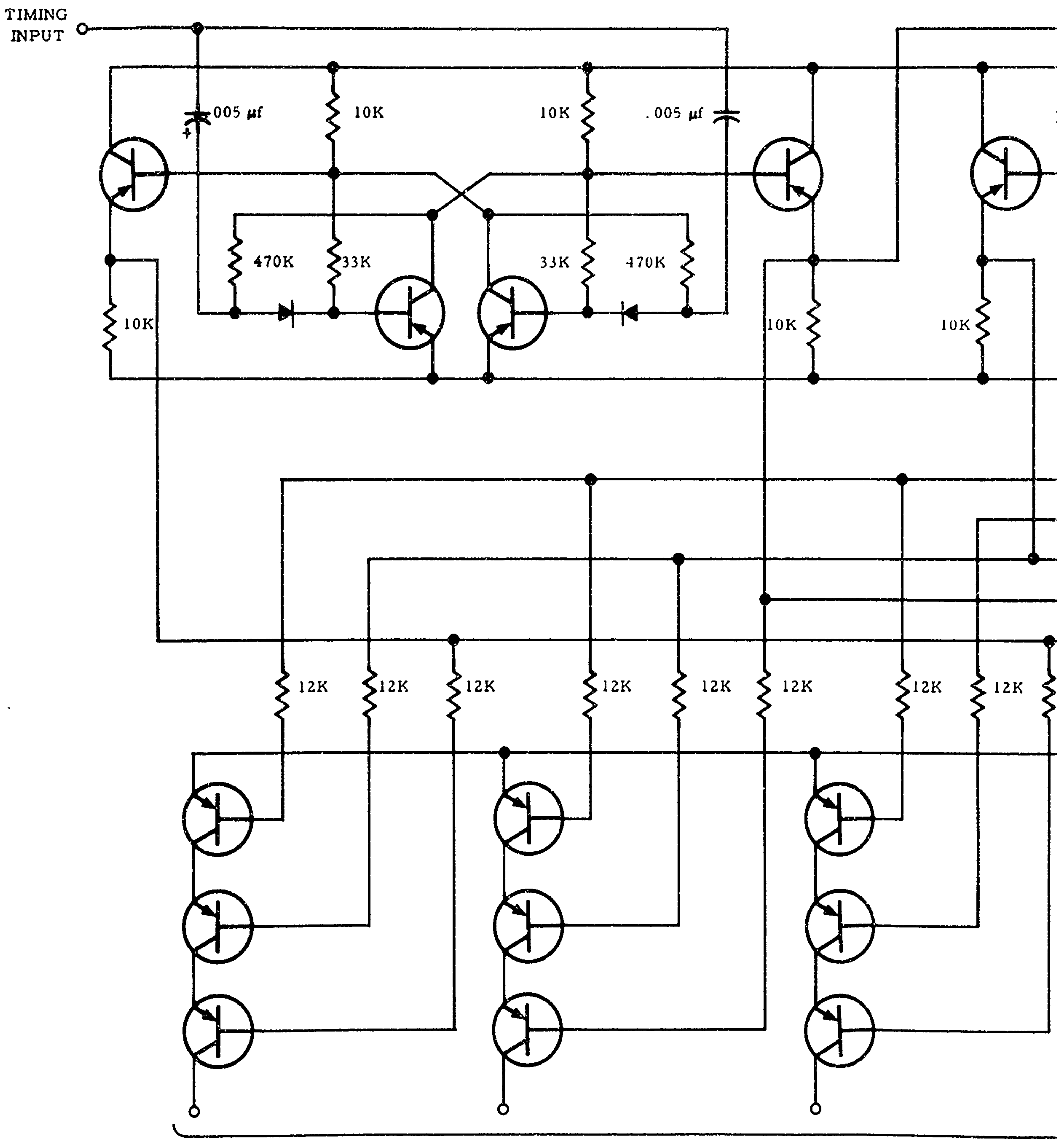
(13) Temperature stability: within ±1.0 per cent of the bandwidth from 9°C to 50°C.

The components for the VCO are on a separate circuit board. (See figure 3-22.) Figure 3-23 is a schematic diagram of the VCO; figure 3-24 is a plot of the VCO input versus frequency output.

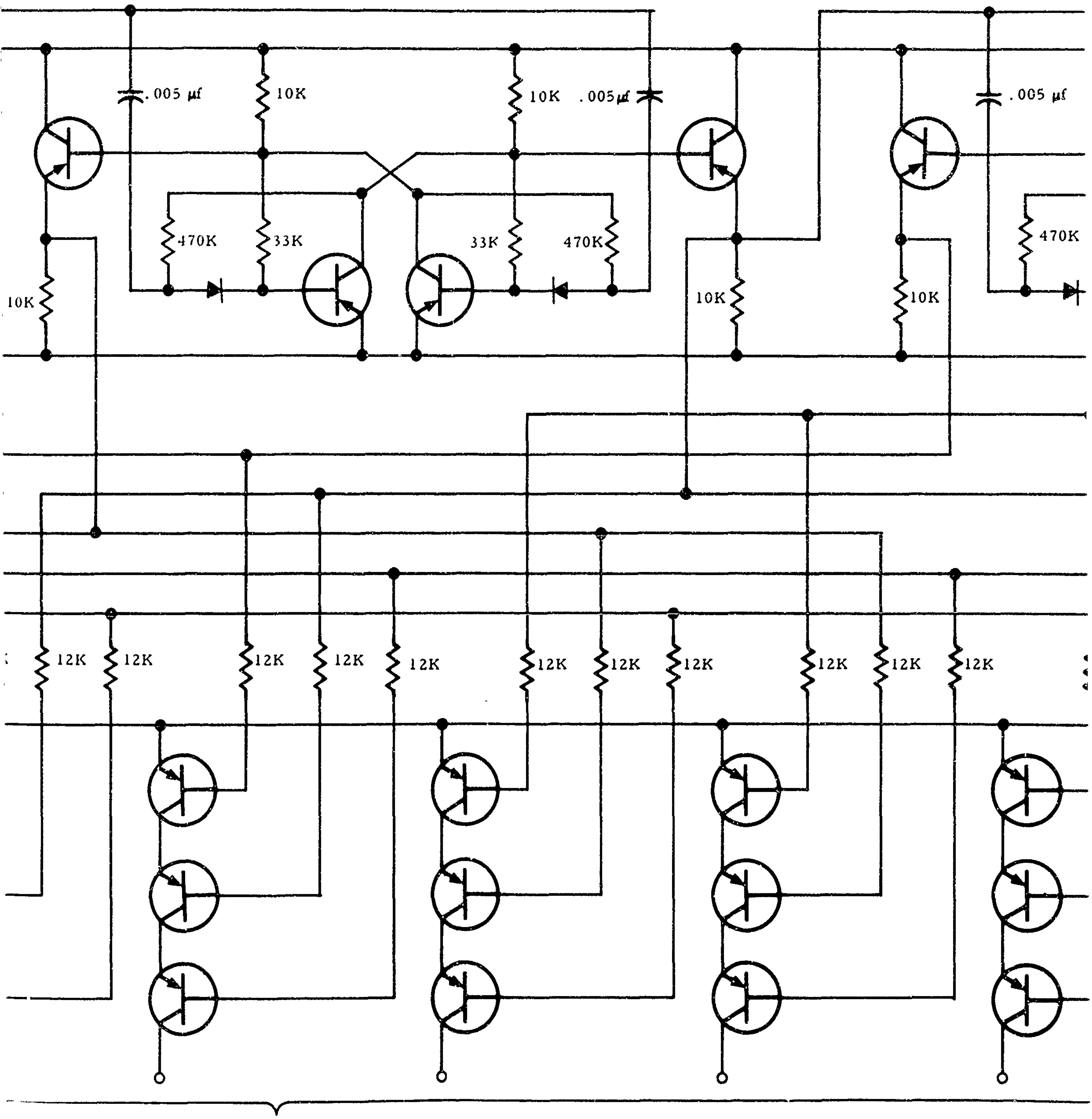
3.6.3.4 Development of Transmitter. The transmitter accepts the output of the VCO and uses it to modulate the rf carrier. The modulated signal is radiated over the telemetry antenna.

(1) Frequency range: 136-137 mc tuned to assigned frequency of 136.11 mc

(2) Modulation: PM

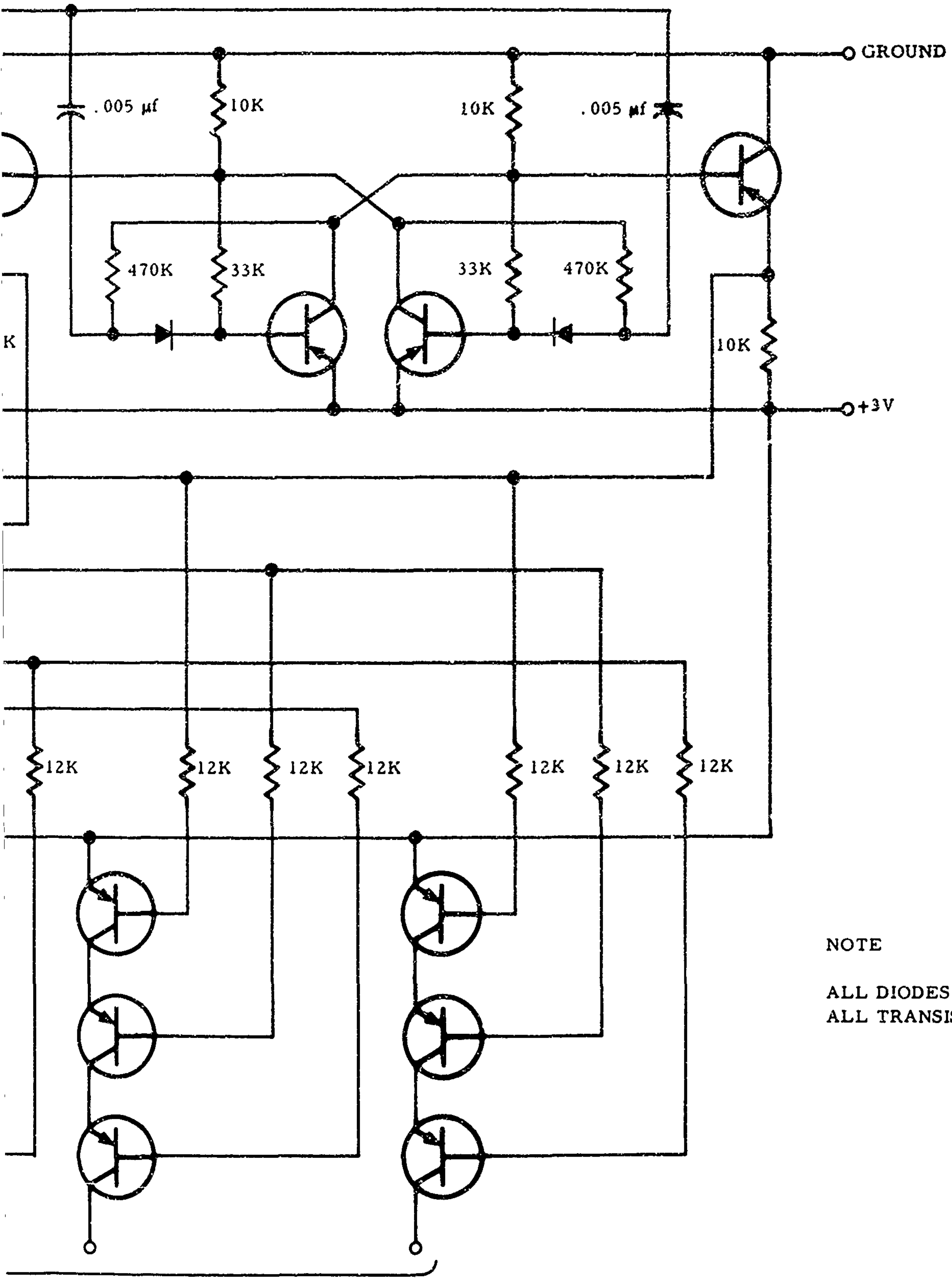

TIMING INPUT
.005 μf
10K
10K
.005 μf
470K
33K
33K
470K
10K
10K
10K
12K
12K
12K
12K
12K
12K
12K
12K

A



TO TIMER BOARD

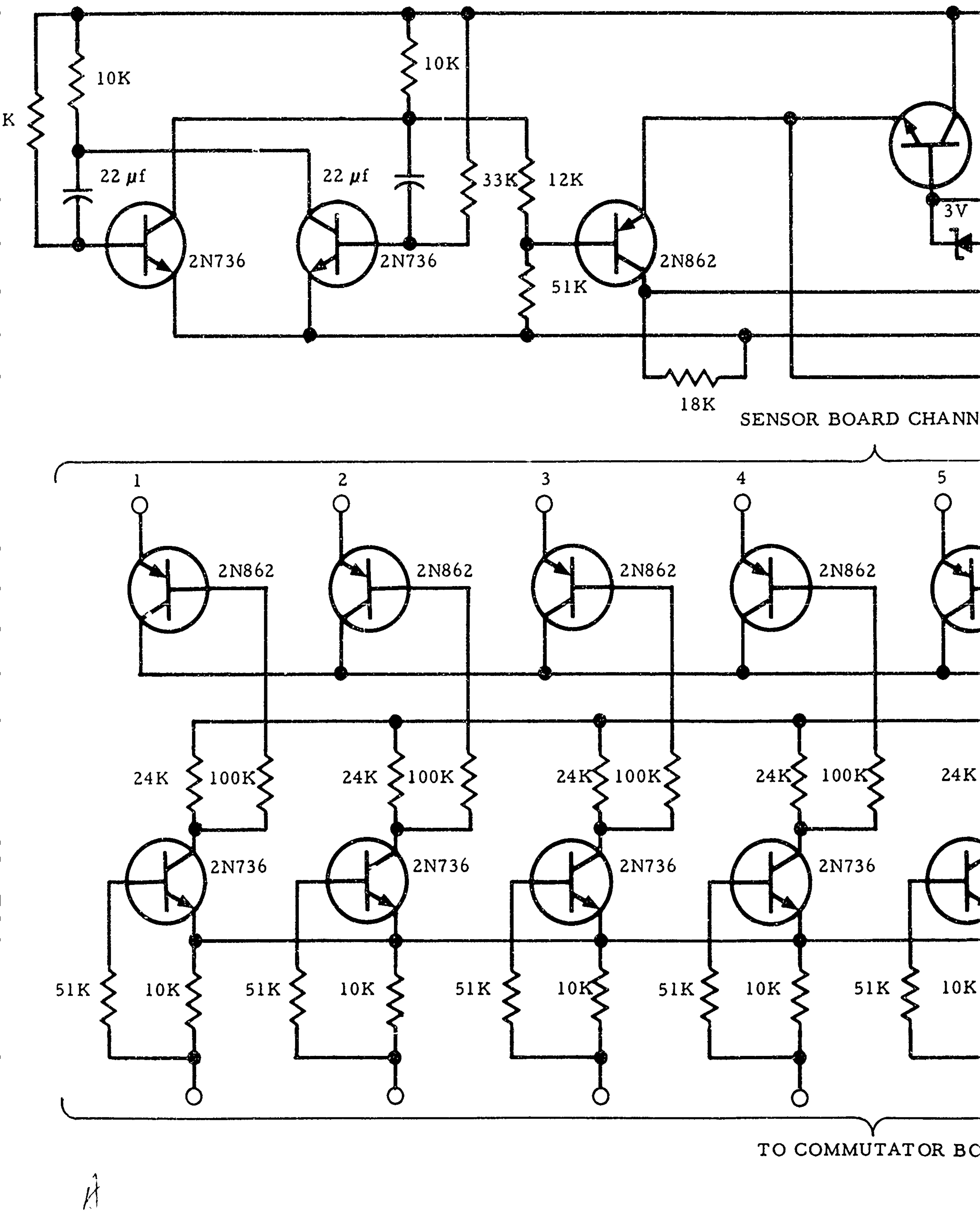
B



NOTE

ALL DIODES ARE 1N662
ALL TRANSISTORS ARE 2N862

Figure 3-20. Schematic Diagram
of the Commutator Board


K
10K
10K
22 µf
22 µf
33K
12K
2N736
2N736
2N862
51K
3V
18K
SENSOR BOARD CHANN
1
2
3
4
5
2N862
2N862
2N862
2N862
24K
100K
24K
100K
24K
100K
24K
100K
24K
2N736
2N736
2N736
2N736
51K
10K
51K
10K
51K
10K
51K
10K
51K
10K
TO COMMUTATOR BO

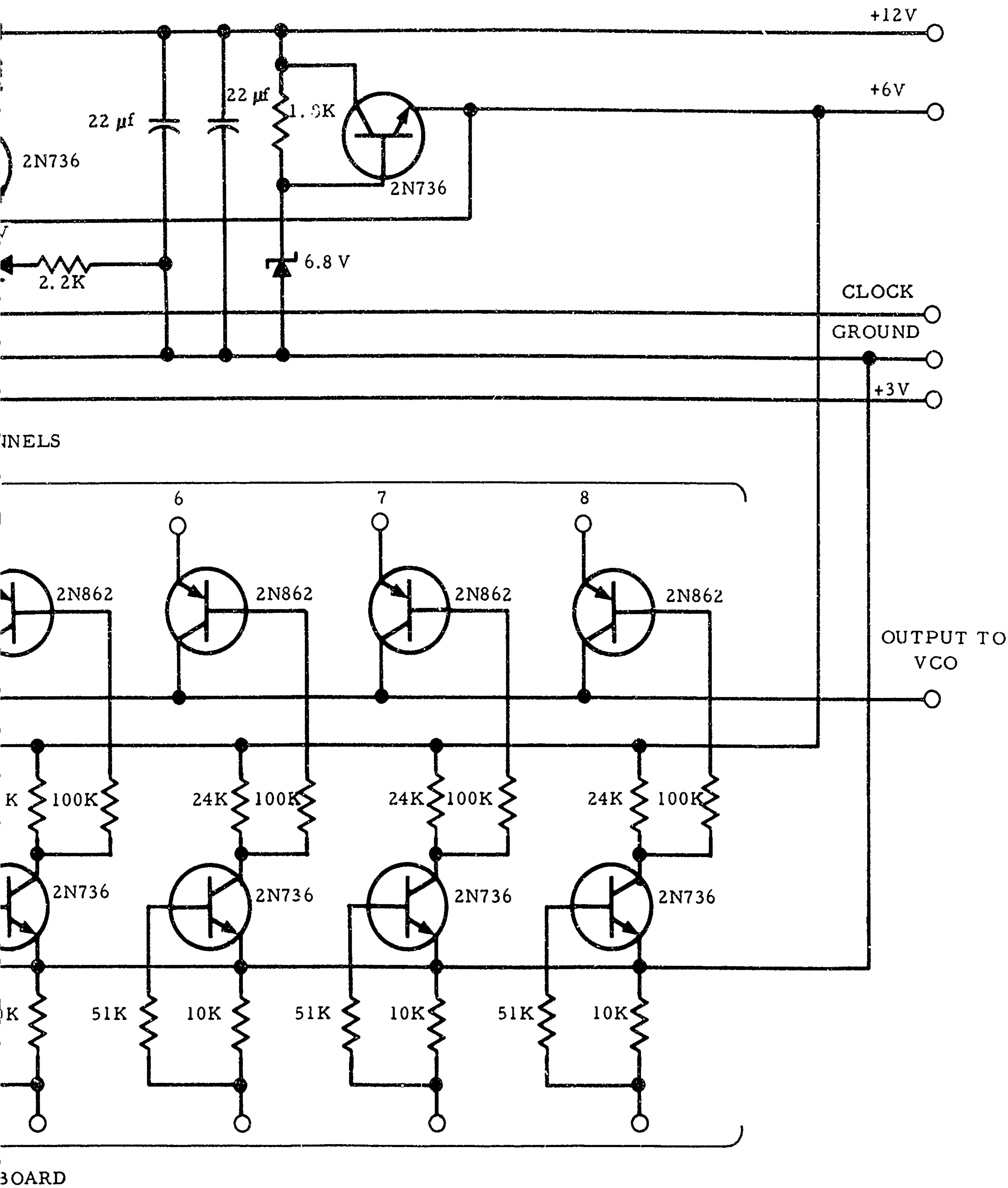
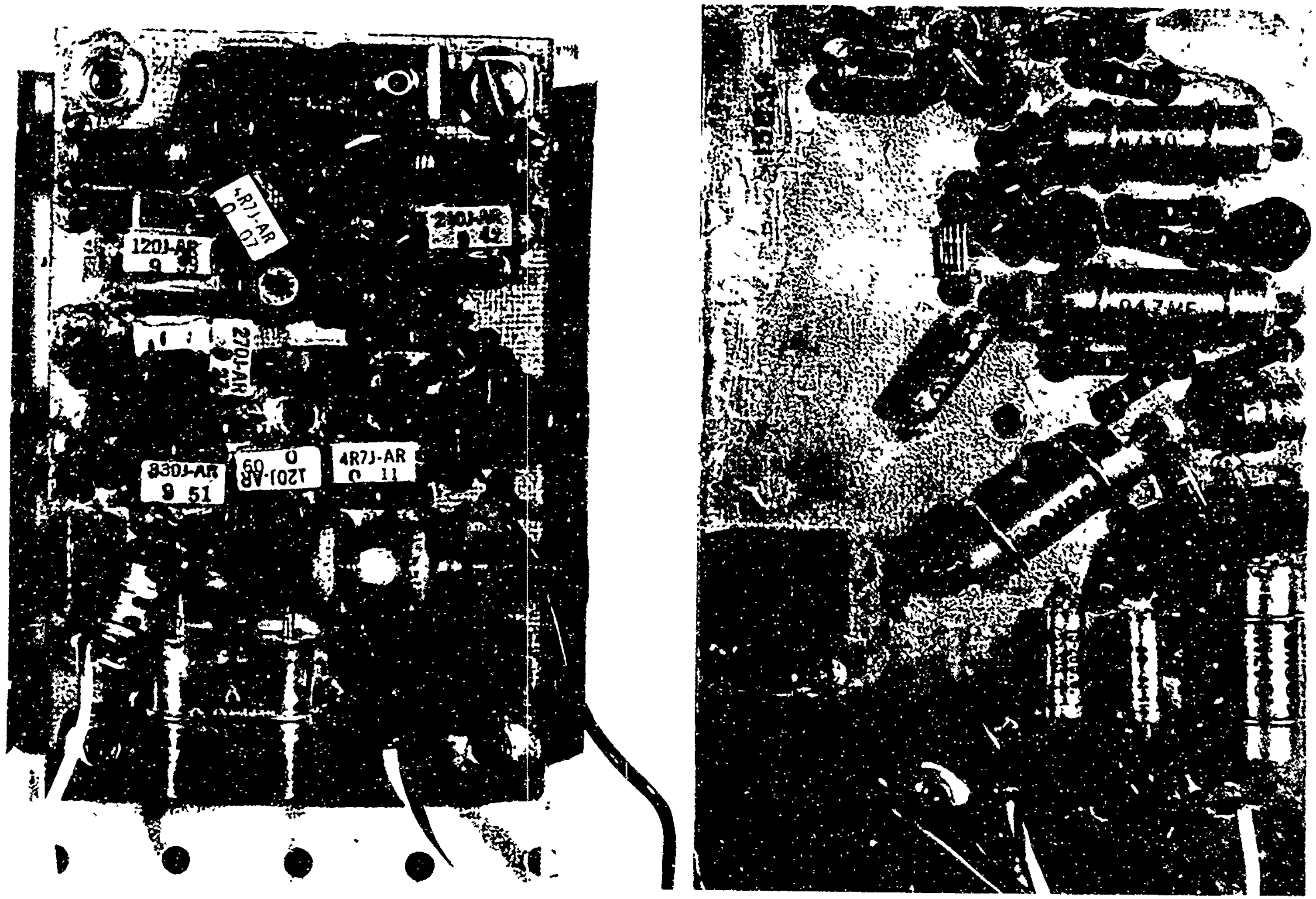


Figure 3-21. Schematic Diagram of the Timer Board



TRANSMITTER VCO

Figure 3-22. Typical Transmitter and VCO Printed Circuit Boards

(3) Modulation frequency range: one subcarrier 730 ±50 cps

(4) Input impedance: greater than 1.0 megohm shunted by 15 $\mu\mu f$

(5) Peak carrier deviation: ±3 kc

(6) Modulation characteristic: 1 radian

(7) Power output: 100 mw in nominal 50-ohm load

(8) Accuracy: carrier frequency accurate to ± 0.01 per cent of assigned frequency

(9) Distortion: total harmonic distortion less than 2 per cent

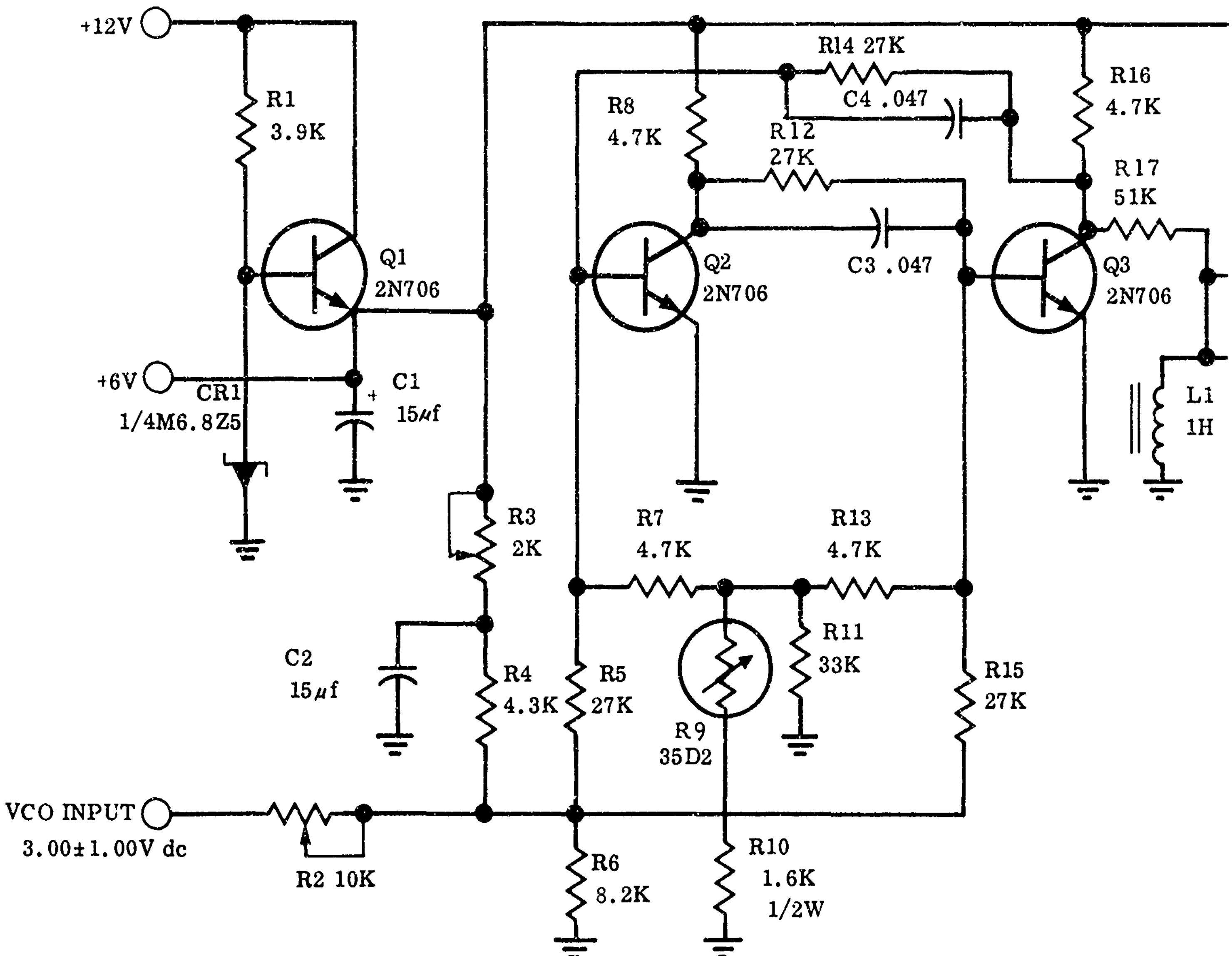
(10) Power requirements: 40 ma @ 12V dc

(11) Life: one year in orbit.

Figure 3-22 and 3-25 illustrate the transmitter printed circuit board and the circuit board cover, respectively. Figure 3-26 is a schematic diagram of the transmitter. Figure 3-27 is a plot of the transmitter frequency variation vs temperature.

3.6.4 Telemetry Subsystem Fabrication. Cubic constructed three telemetry subsystems. All materials and compone ts were selected from the latest revision of the Qualified Parts List (QPL). Workmanship has been in accordance with the highest standards applied to MIL-Specifications hardware. In view of the need for high reliability and long life, all stresses (electrical, magnetic, thermal, and mechanical) have been kept as low as the requirement for compactness and lightweight permits. Each subsystem is interconnected as shown in figure 3-28.

3.6.5 Telemetry Subsystem Checkout and Calibration. After fabrication each subsystem was tested and calibrated. Tests consisted of an adjustment of the transmitter modulation index and



+12V
R1
3.9K
Q1
2N706
+6V
CR1
1/4M6.8Z5
C1
15μf
R3
2K
C2
15μf
R4
4.3K
VCO INPUT
3.00±1.00V dc
R2 10K
R14 27K
C4 .047
R8
4.7K
R12
27K
Q2
2N706
C3 .047
R16
4.7K
R17
51K
Q3
2N706
L1
1H
R7
4.7K
R13
4.7K
R5
27K
R9
35D2
R11
33K
R15
27K
R6
8.2K
R10
1.6K
1/2W

A

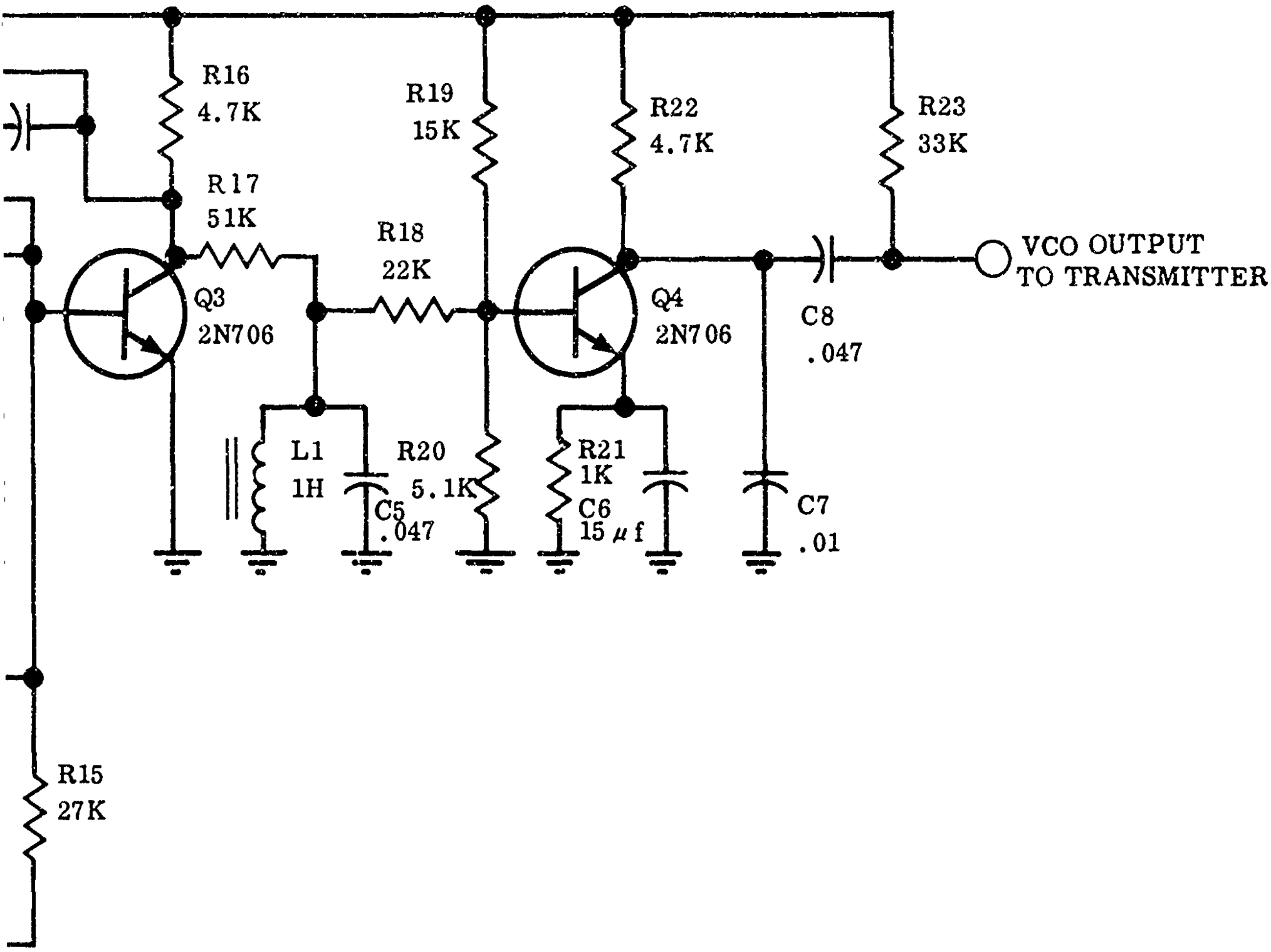


Figure 3-23. Schematic Diagram of the VCO

13

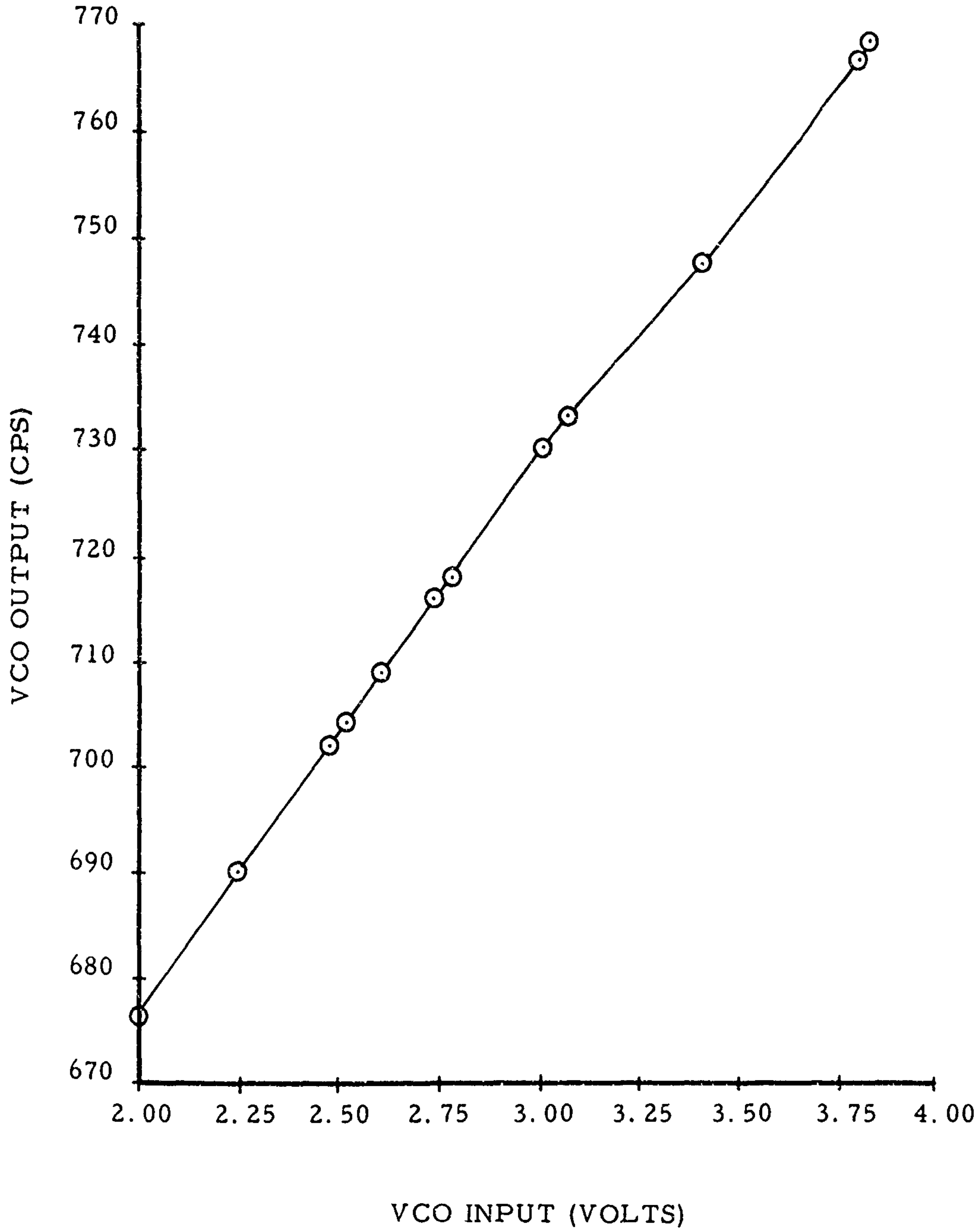


Figure 3-24. VCO Response Frequency VS Input Voltage

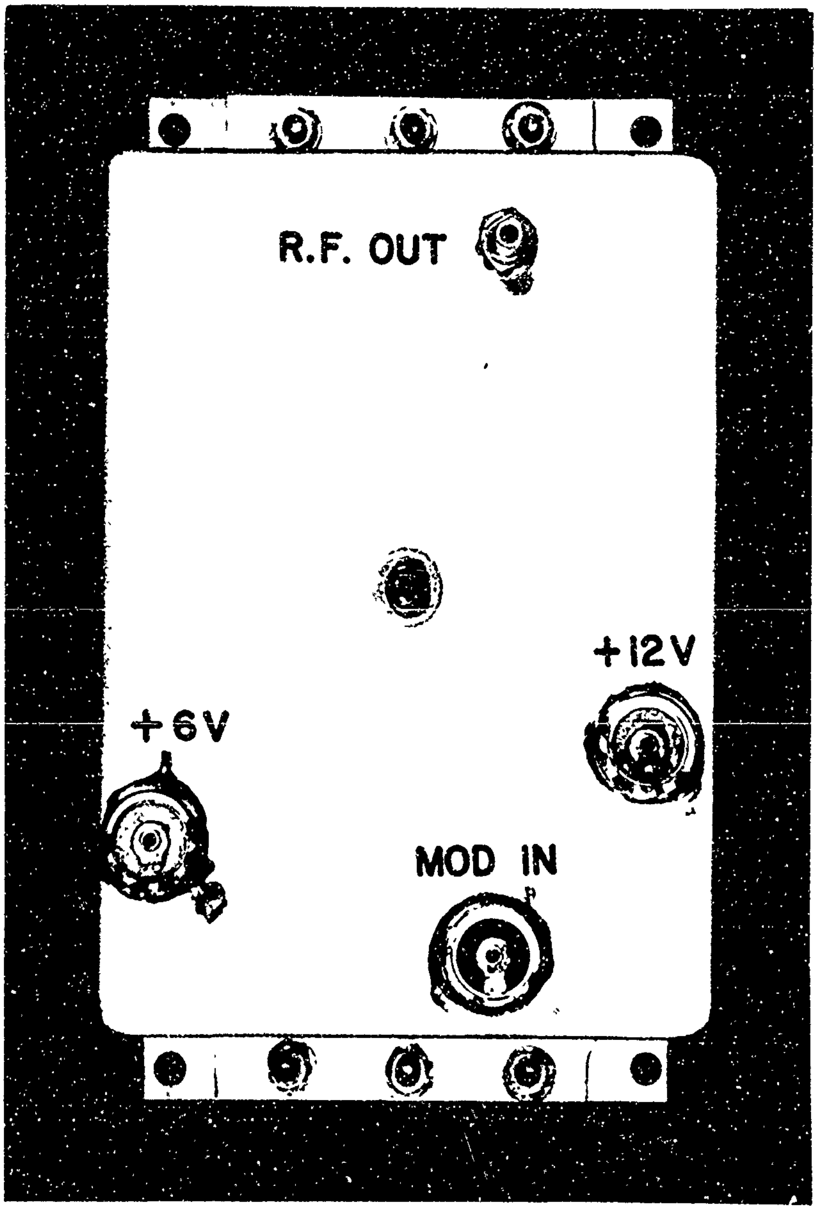
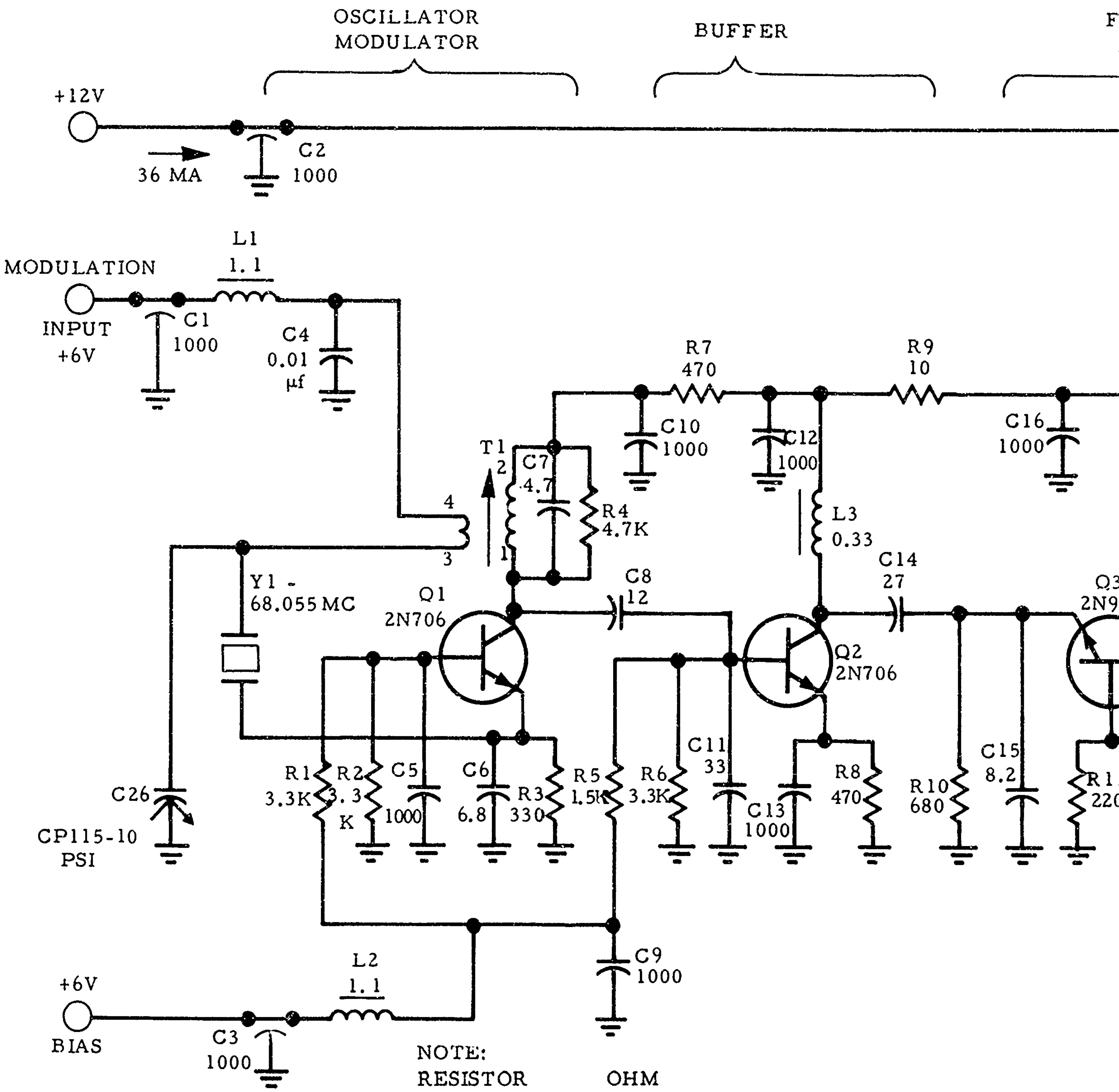


Figure 3-25. Typical Transmitter Cover



NOTE:
RESISTOR OHM
CAPACITORS µµf
INDUCTORS µh
UNLESS INDICATED OTHERWISE

A

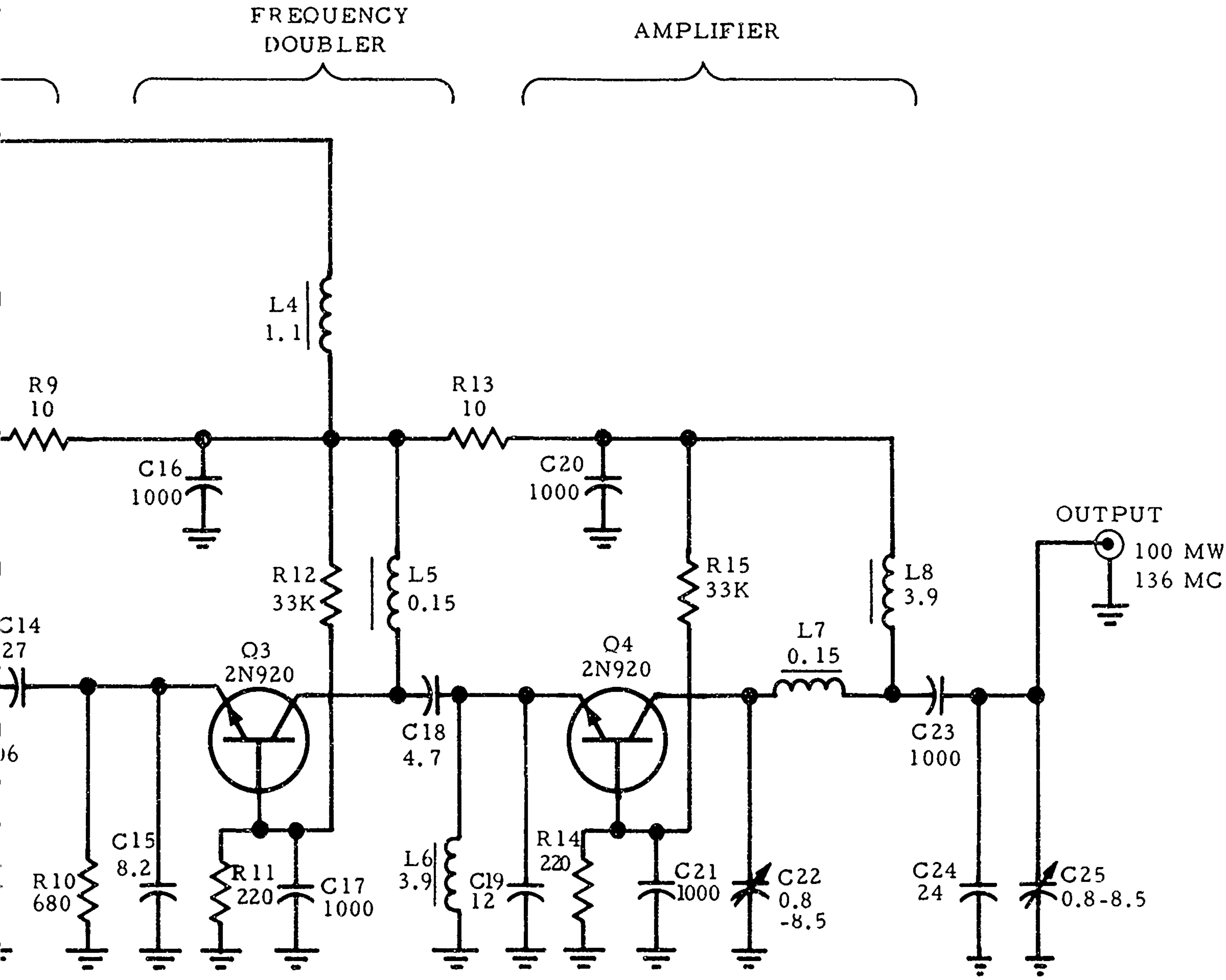


Figure 3-26. Schematic Diagram of Telemetry Transmitter

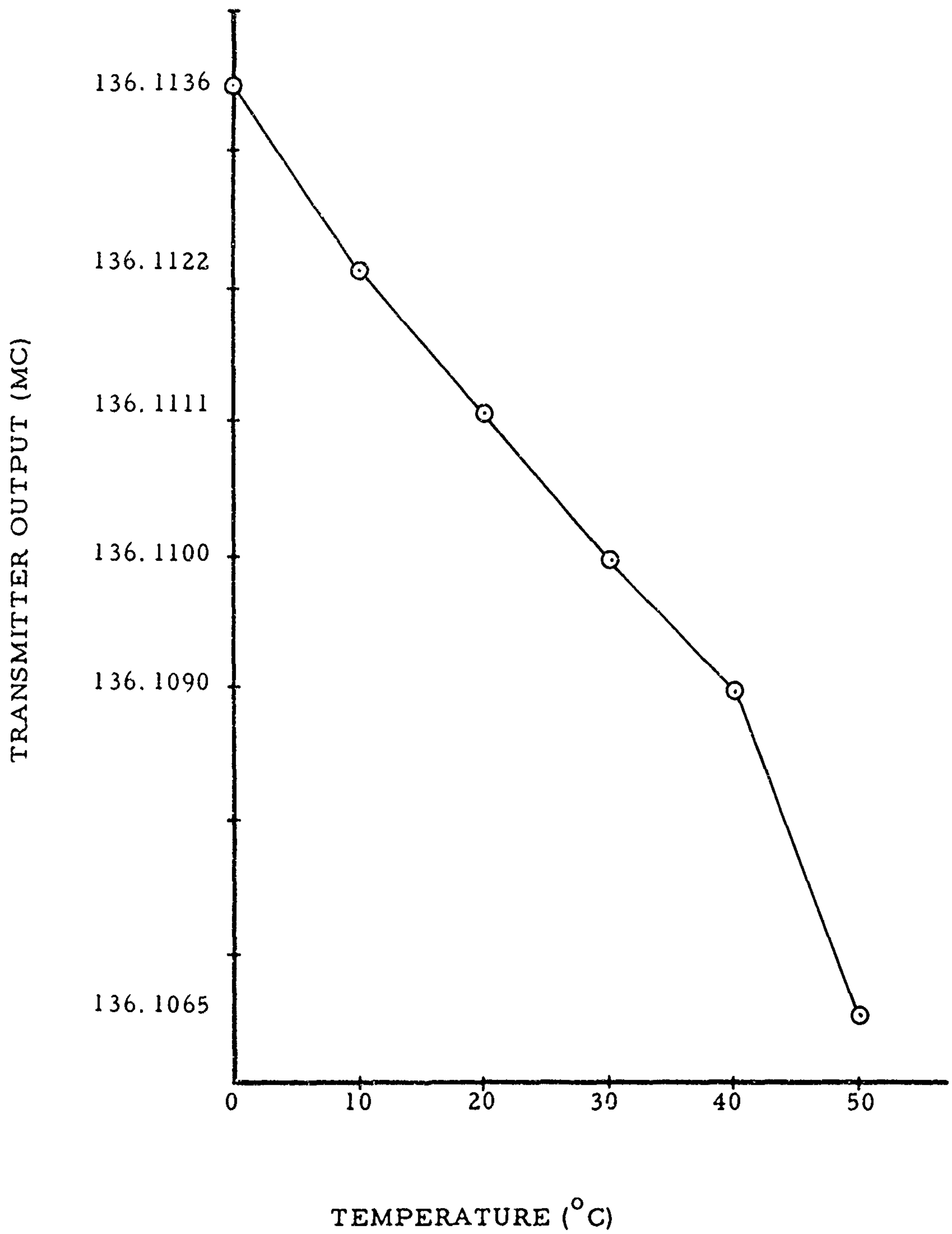


Figure 3-27. Transmitter Frequency Variation VS Temperature

environmental tests. A discussion of the environmental test is covered in a later section. Table IV is a list of the test equipment utilized to checkout and calibrate the telemetry subsystem.

3.6.5.1 Adjustment of Transmitter Modulation Index. To adjust the transmitter modulation index, the test setup shown in figure 3-29 was used. The test oscillator and signal generator were used to establish a first-order sideband reference of unity. Telemetry was then connected to the receiver and the circuit elements were varied until first-order sidebands of unity were achieved at a power output of 100 mw. (An FM modulation index of unity corresponds to a PM modulation index of 1 radian.)

3.6.5.2 Calibration. Figure 3-30 shows the test setup used to calibrate the telemetry subsystem. The following measurements were made:

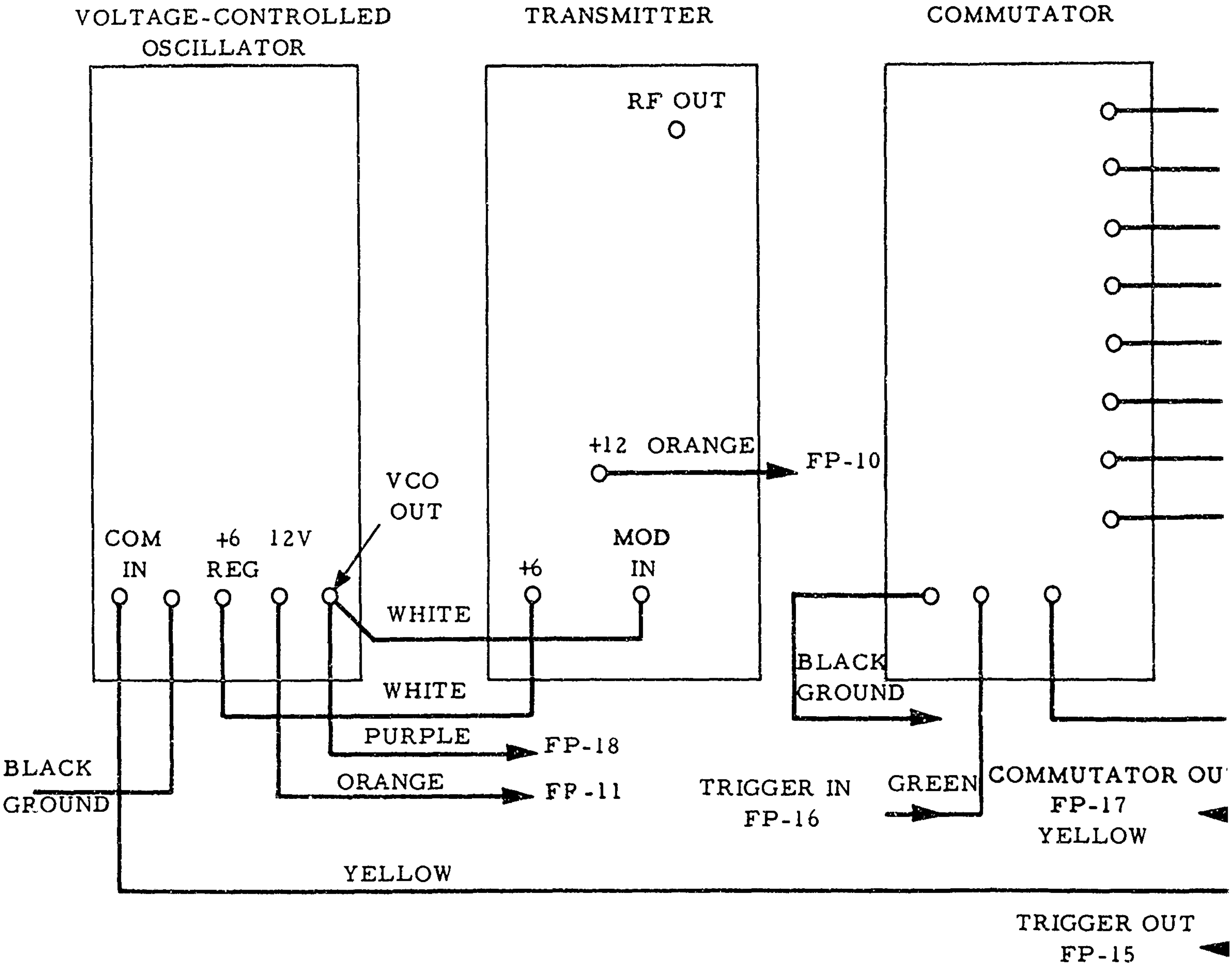
(1) VCO output vs battery voltage (figure 3-31).

(2) VCO output vs transponder plate current (figure 3-32).

(3) VCO output vs transponder plate voltage (figure 3-33).

Figures 3-34 and 3-35 are composites of the channel outputs vs temperature. Note the nonlinearity of channel 2 in figure 3-34. Adjustment of the 3V and 6V regulator to provide more current drain flattened the curve as shown in figure 3-35. Figure 3-36 shows typical commutator channel outputs taken at a temperature of 30°C. Refer to paragraph 3.19 for actual recorded strip-chart.

3.7 Developmental History of the Power Supply Subsystem. The power supply subsystem consists of the solar cell plaques, storage batteries, and the charging network. Cubic determined the solar cell and storage battery requirements and designed the charging network. Hoffman Electronics Corporation and International Rectifier Corporation constructed the solar cell plaques, and the storage batteries were purchased from Sonotone Corporation.


VOLTAGE-CONTROLLED OSCILLATOR
TRANSMITTER
COMMUTATOR
RF OUT
+12 ORANGE
FP-10
VCO OUT
COM IN
+6 REG
12V
+6
MOD IN
WHITE
WHITE
PURPLE
FP-18
ORANGE
FP-11
BLACK GROUND
BLACK GROUND
TRIGGER IN FP-16
GREEN
COMMUTATOR OU
FP-17
YELLOW
YELLOW
TRIGGER OUT FP-15

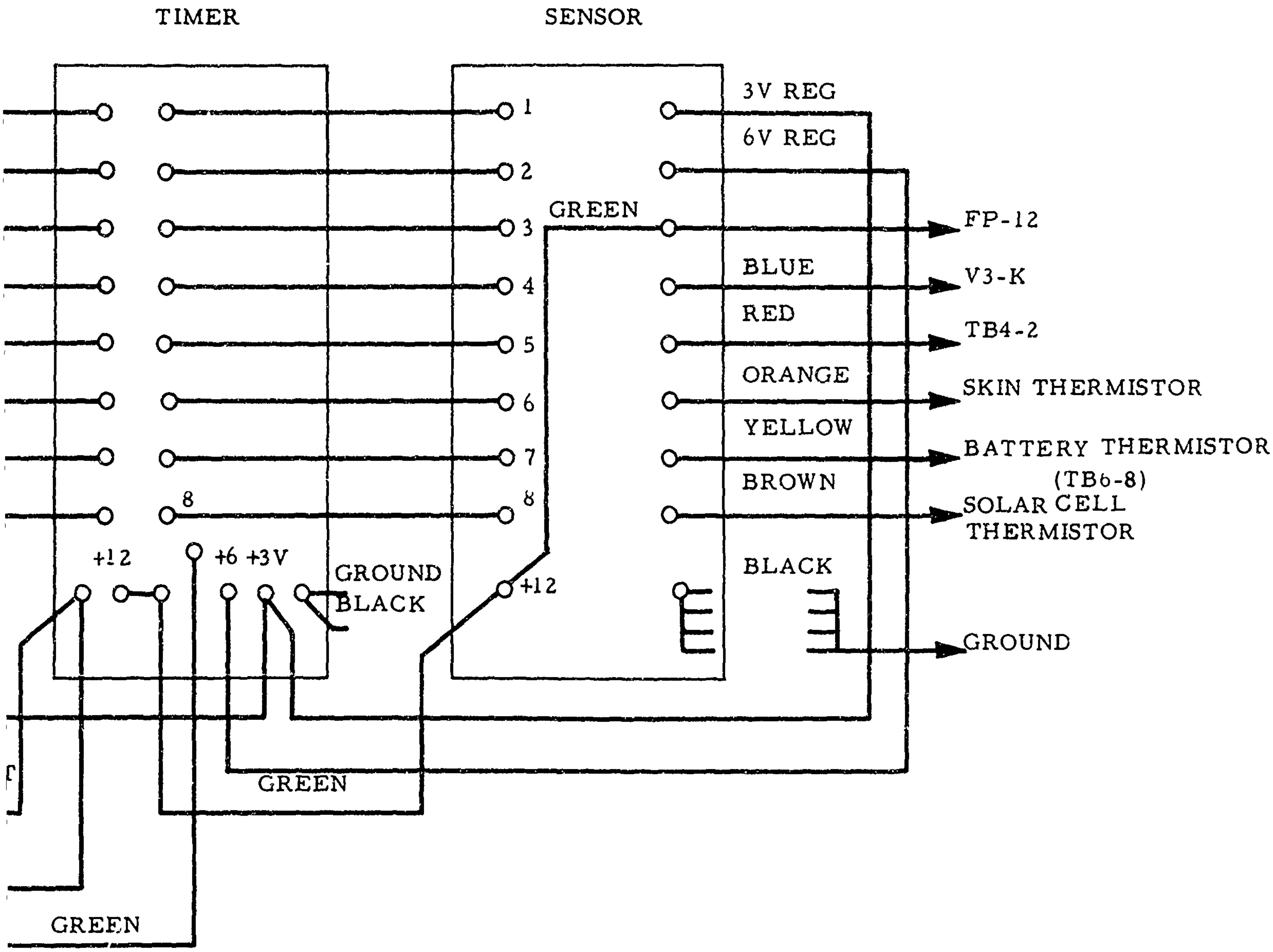


Figure 3-28. Telemetry Subsystem Interconnections

TABLE IV

TEST EQUIPMENT UTILIZED
FOR
TELEMETRY CHECKOUT AND CALIBRATION

Description	Manufacturer	Part No.
Oscillator	Hewlett-Packard	200CD
Power meter	Hewlett-Packard	430C
FM-AM signal generator	Boonton Radio	202E
Receiver	Eddystone	770R
Oscilloscope	Tektronix	546 with type L plug-in unit
Frequency counter	Beckman/Berkely	7370
Oven	Delta Design	7000A
DC voltage source	Kepco	800

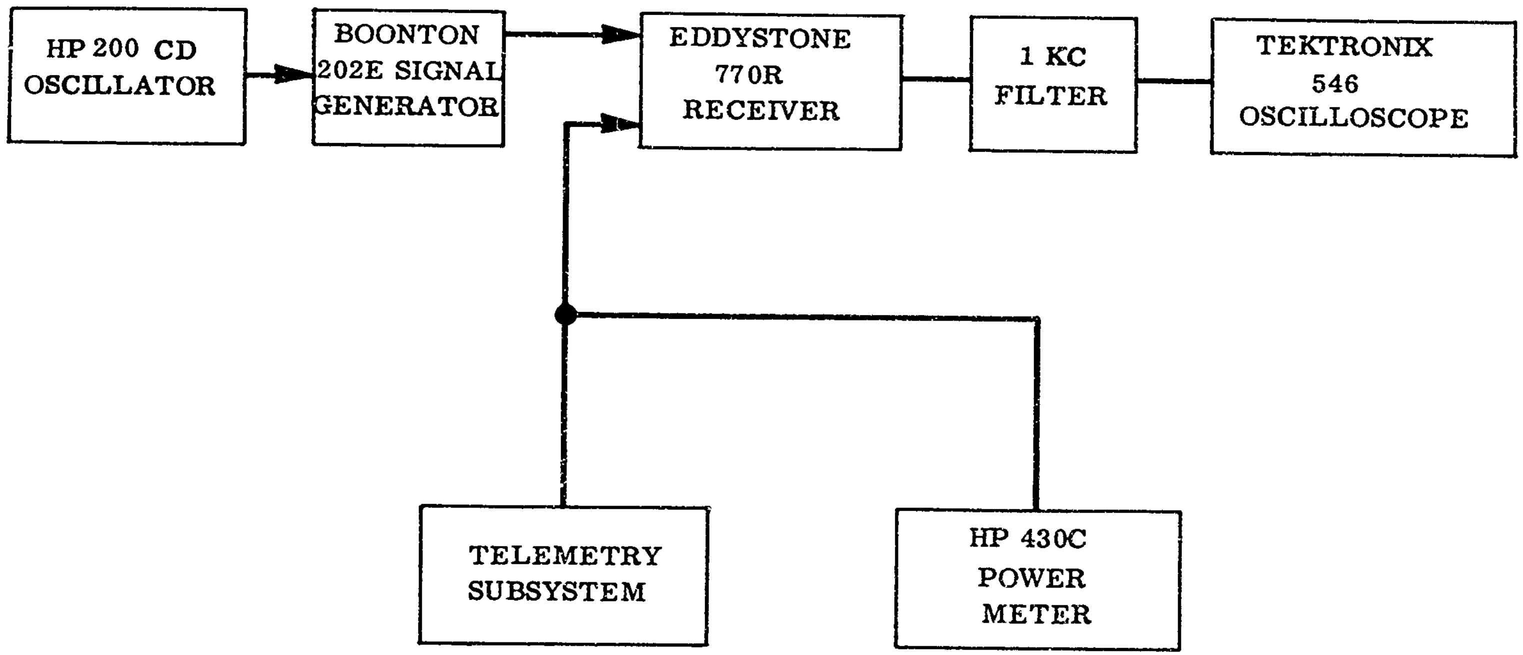


Figure 3-29. Test Setup for Adjustment of the Telemetry Modulation Index

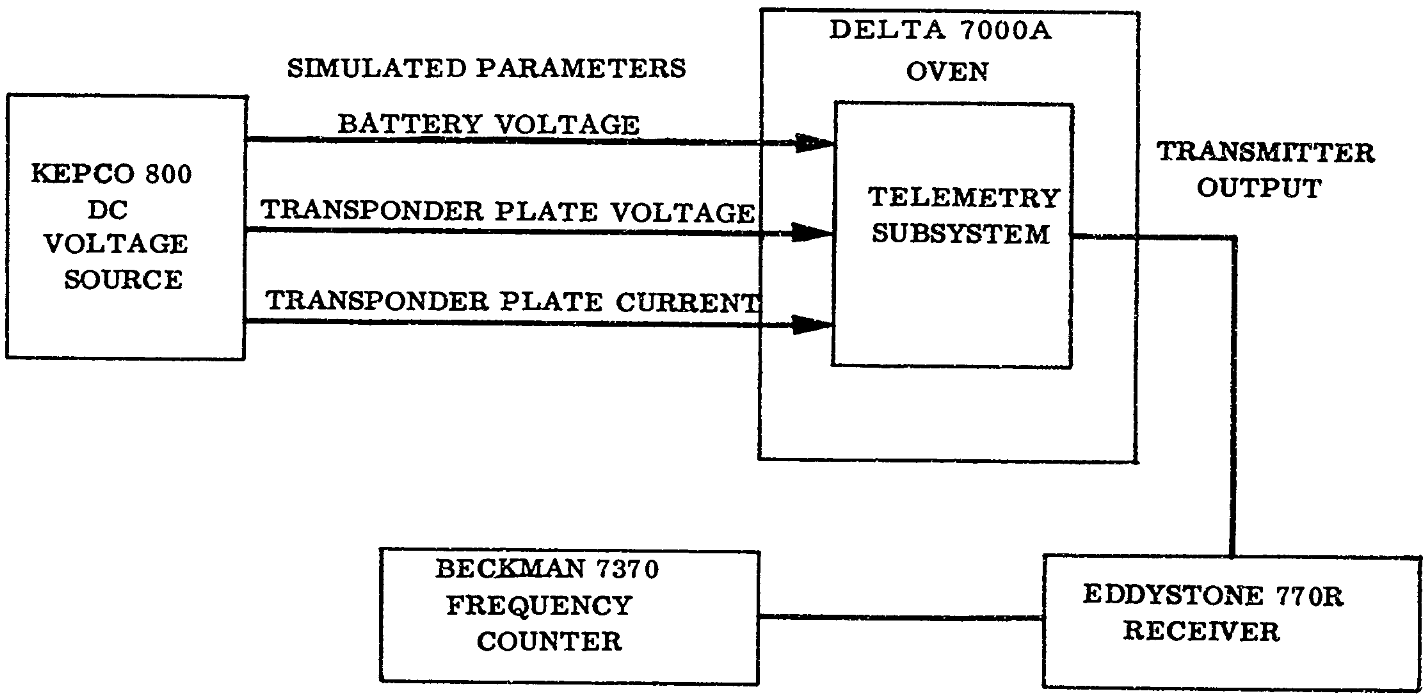


Figure 3-30. Test Setup for Calibration of the Telemetry Subsystem

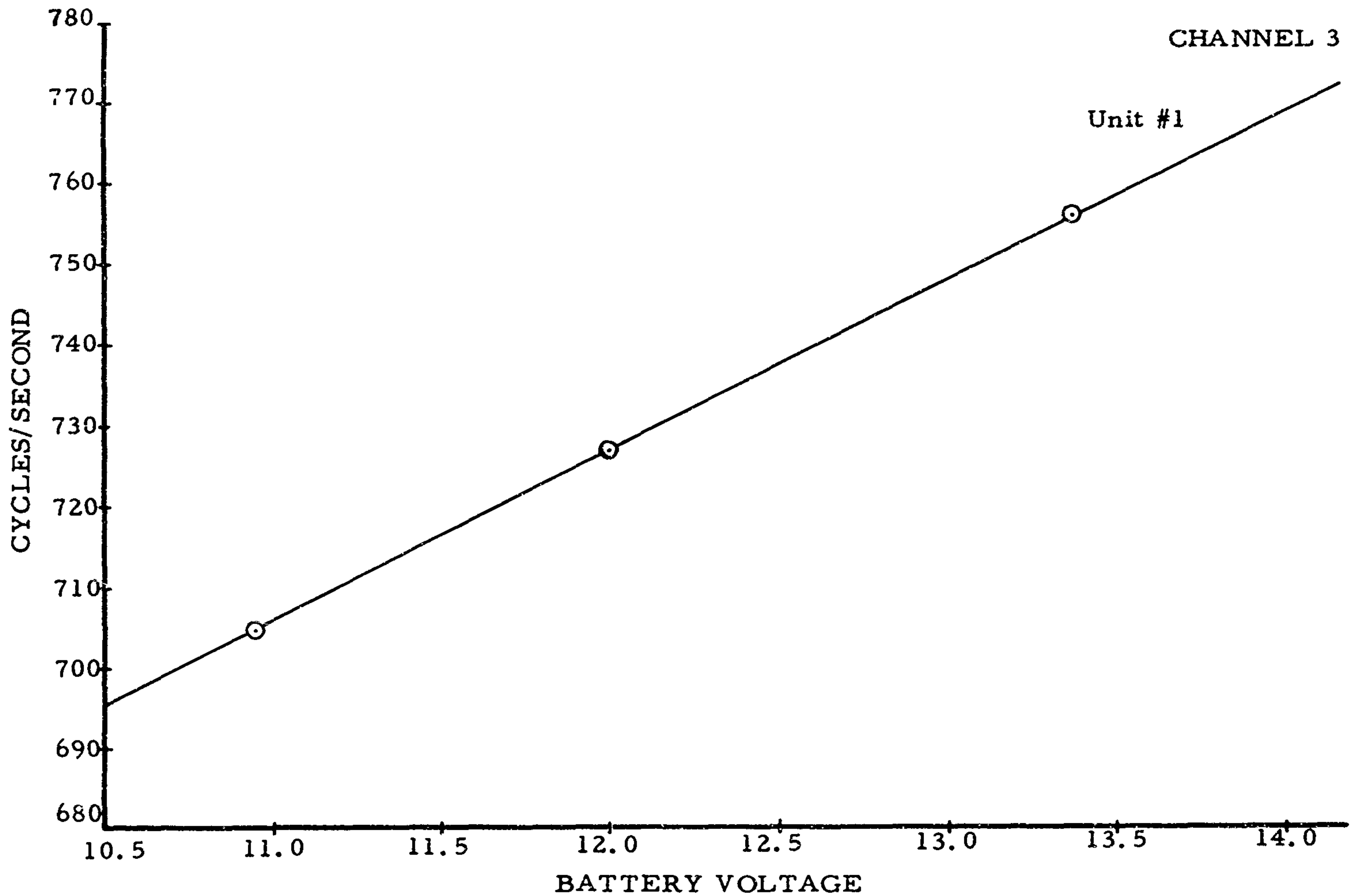


Figure 3-31. Typical VCO Output VS Battery Voltage

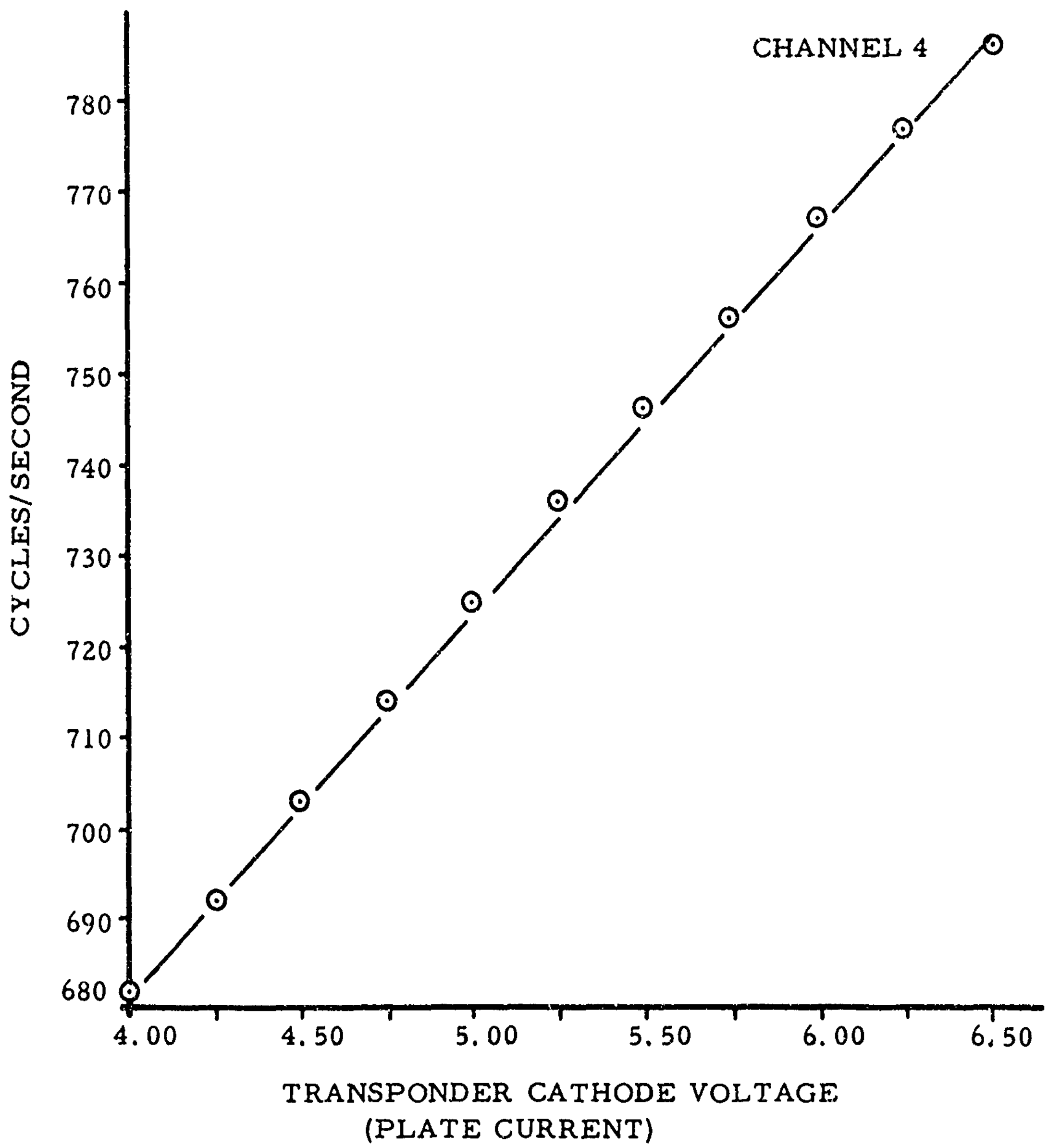


Figure 3-32. Typical VCO Output VS Transponder Cathode Voltage (Plate Current)

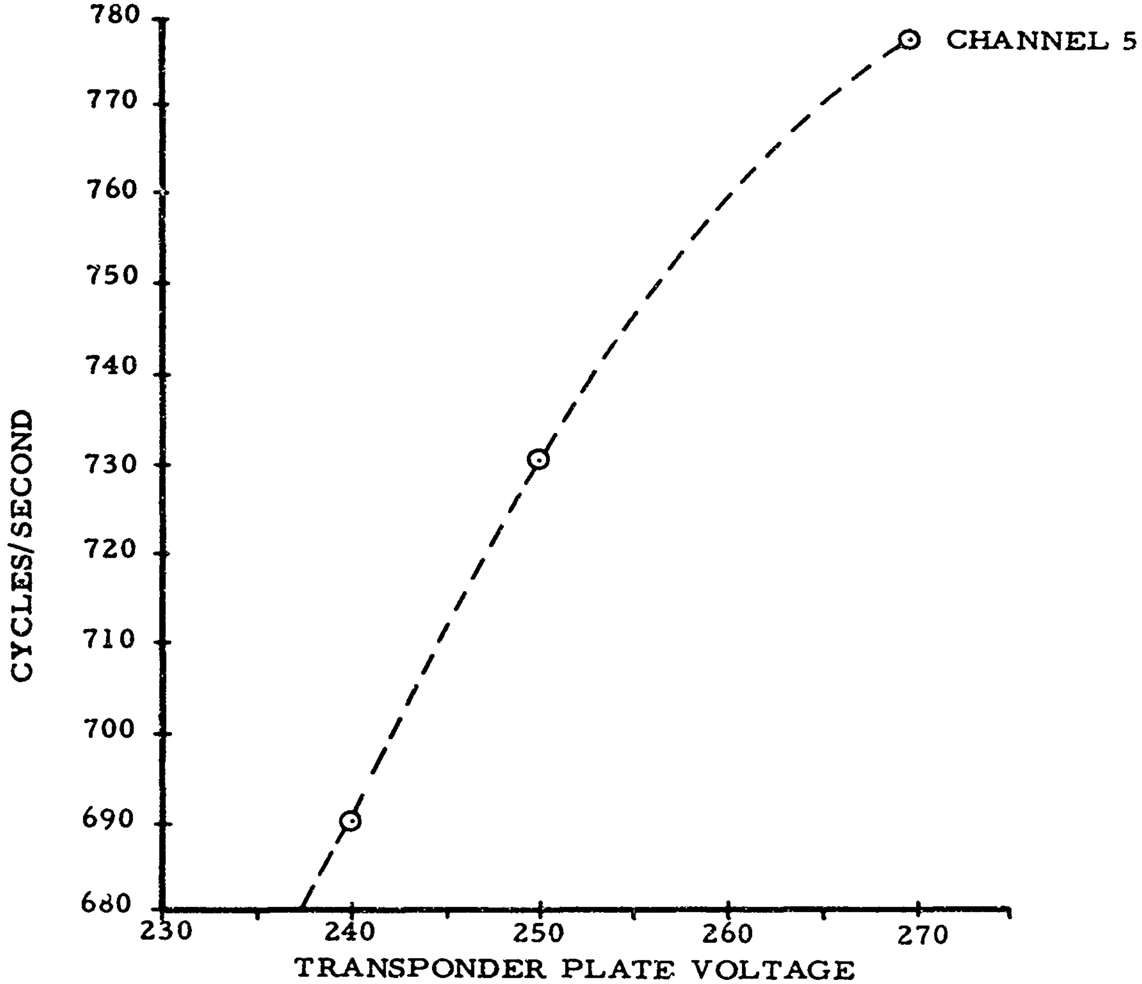
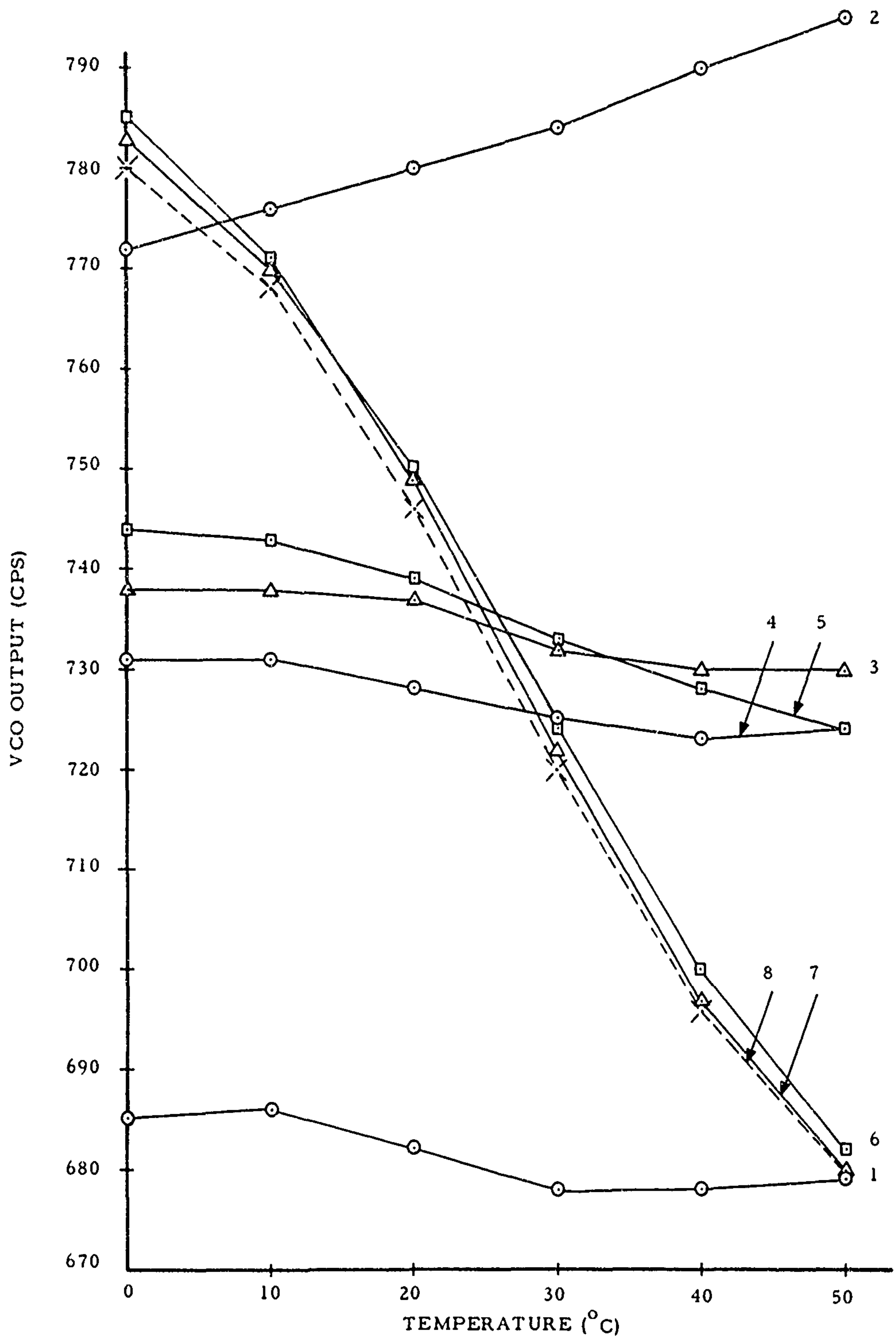


Figure 3-33. Typical VCO Output VS Transponder Plate Voltage



3-35

Figure ~~3-34.~~ Typical Telemetry Channel VCO Outputs
VS Temperature (~~Before~~ After Adjustment)

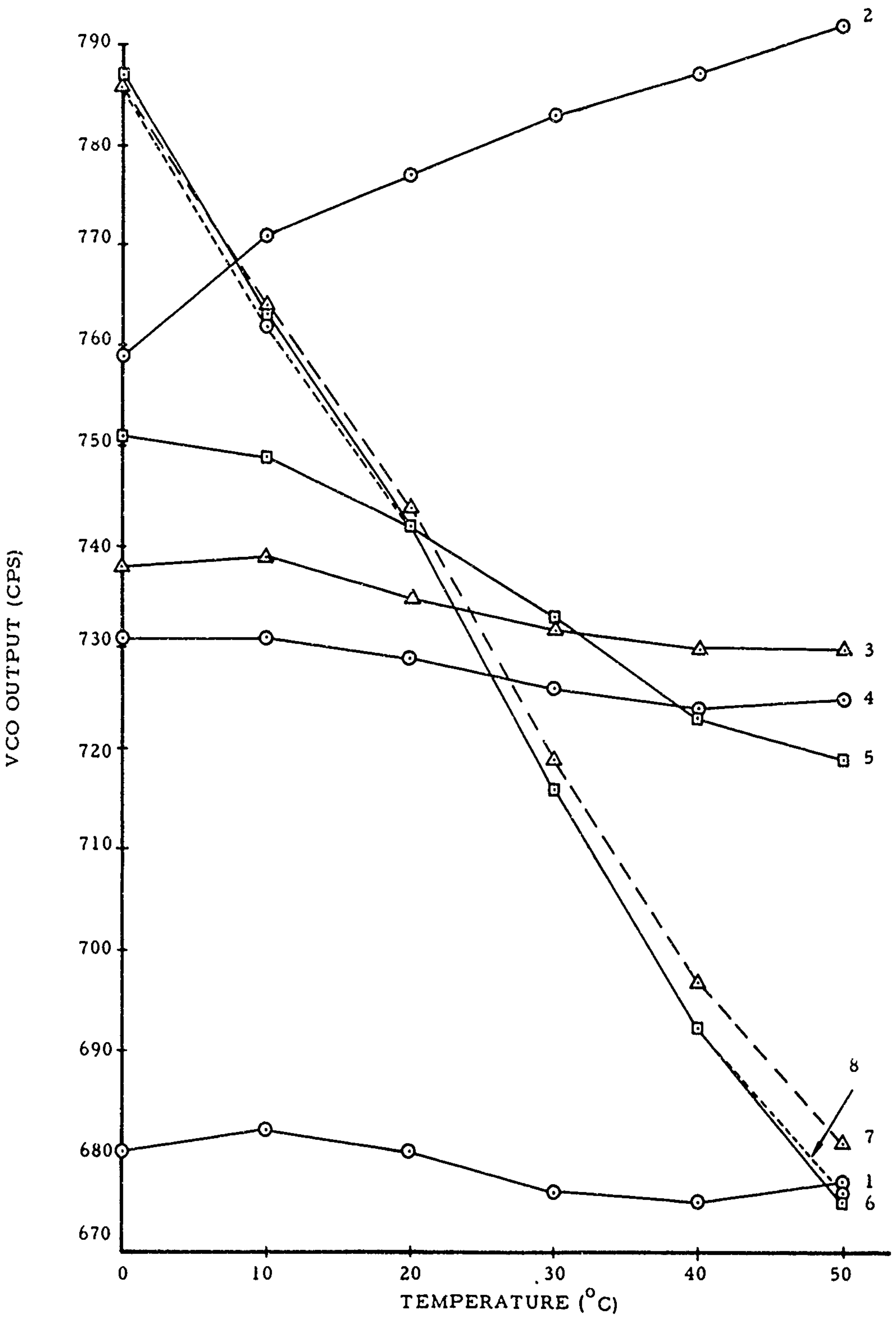
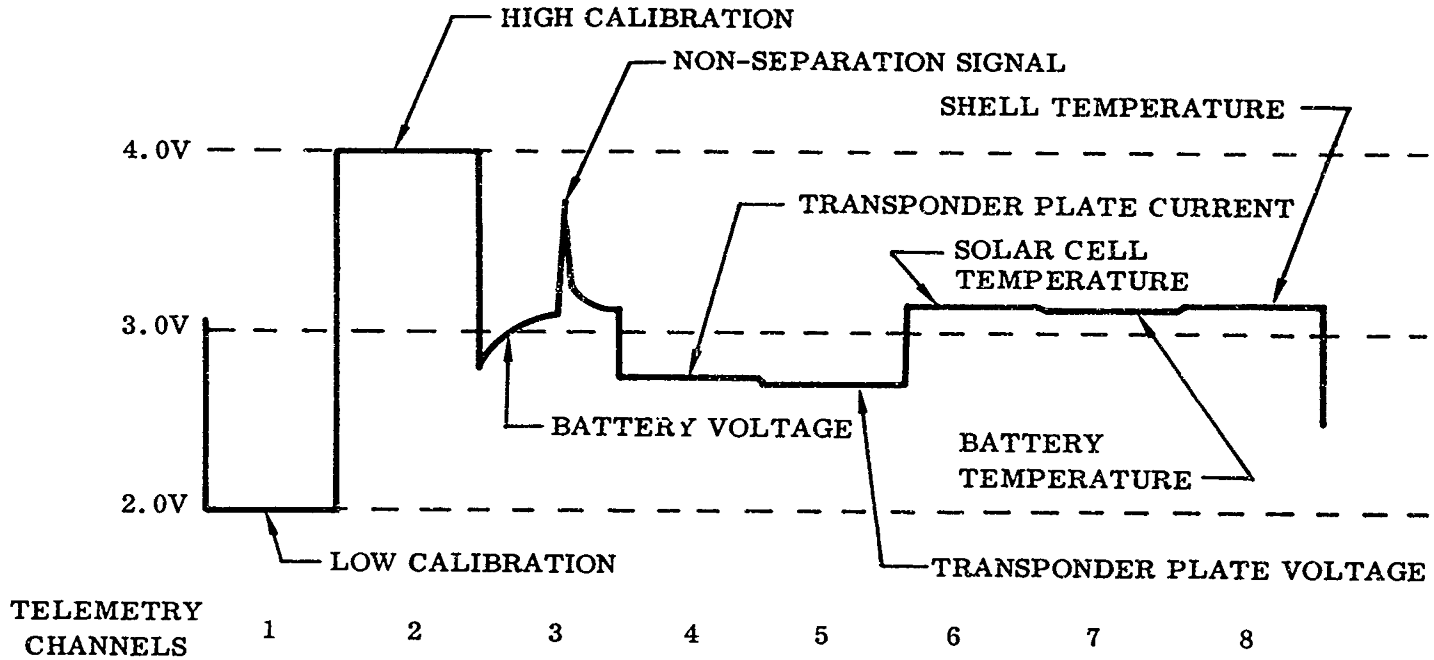


Figure ~~3-35~~ 3-34. Typical Telemetry Channel VCO Outputs VS Temperature (~~After~~ Before Adjustment)



NOTE - TRANSPONDER PLATE CURRENT AND VOLTAGE INDICATIONS ARE SHOWN AT LEVELS TYPICAL DURING SECOR OPERATION

Figure 3-36. Typical Commutator Output

3.7.1 General Problem. The general problem in the design, fabrication and installation of the power supply subsystem involved the development of an integrated power source for the spacecraft electronics and a suitable recharginig network for that source.

3.7.2 Preliminary Considerations.

3.7.2.1 Power Supply Loads. Three loads are to be supplied by the solar-cell/storage-battery system, two intermittent and one substantially constant, as follows:

(1) The TR-17 in transponder configuration, requires 2.0 amperes at 12V. It was desired to provide a capability of one hour per 24-hour day, at least for days in which the orbits are 90 per cent sunlit or more. A reduced period of daily operation can be tolerated for days in which the orbits are less than 90 per cent sunlit, down to a minimum of about 45 minutes per day for the minimum (63 per cent) sunlit orbit. Thus there is a maximum requirement of 2.0 ampere-hours per day.

(2) The monitor transmitter, telemeters "housekeeping" aboard the spacecraft and will consume about 0.050 ampere at 12V during operation. It was planned to have this turned on at the time of launch and subsequently turned off or on as required using additional channels of the TR-17. Thus, although this is an intermittent load in the sense that it can be turned off, the possibility exists that it may be on for several days or weeks at a time, and the solar-cell/storage-battery system had to be planned accordingly.

(3) The TR-17, in standby configuration, requires a steady 0.028 ampere at 12V. The maximum continuous load is then 0.078 ampere, the minimum load is 0.028 ampere.

3.7.2.2 Suitable Storage Batteries. The only spacecraft-proven storage battery is the nickel-cadmium sealed-cell battery in which the sealing is effected by fused-glass terminal insulation and heliarc container welding. Any less comprehensive sealing will eventually result, under prolonged space-vacuum conditions, in the leakage of some cells and complete inability of the storage battery to operate. The "WS" series of nickel-cadmium cells, as manufactured

by Sonotone, it the only one so qualified at this time. As a further preclusion against leakage, Cubic Corporation has proposed to encapsulate the storage battery system in order to provide a pressurized seal.

3.7.2.3 To provide a 12V system with ~~allowances for isolating diode drops with~~ nickel-cadmium cells, it was necessary to series-connect ten such cells, to form one "bank". To ~~assure~~ insure reliable operation, two such banks are ~~provided~~ connected in parallel, each bank can handle the entire intermittent load ~~with provisions for parallel operation normally, or individual operation of one bank in in the event of failure of the other bank~~. Thus, a total of 20 cells is required. (See figure 3-37.)

3.7.2.4 Each cell is a type WS-103, Sonotone P/N 22902. These are rated in excess of three ampere-hours at the 5-hour rate. They can provide 2.6 ampere-hours at the 1-hour rate. Thus, operation on the two parallel banks will result in a 40 per cent depth of discharge and on a single bank about 80 per cent (It is not necessary to adhere to the 10 per cent depth of discharge employed in some other spacecraft which, because of their assigned mission, impose more severe requirements on the storage battery system.)

Twenty WS-103 cells weigh 6.7 pounds and require a height of 12 inches in the central cylindrical tube; thus a grouping was made of four cells in square array stacked five high.

3.7.2.5 Suitable Solar Cells. The silicon junction diode, an outgrowth of the semiconductor antenna, has been employed exclusively in spacecraft solar energy converters. The most widely used is Hoffman Electronics Corporation Type 120CG, which was planned for use in the Geodetic Spacecraft. Although these have been made with efficiencies as high as 14 per cent conservative design practice indicates that an average efficiency of 12 per cent should be anticipated (Hoffman Type 120CG-12). Each such cell produces 0.485V under exposure to "one sun". To charge the nickel-cadmium cells, a voltage of 1.45V per cell is required. Making allowances for voltage drops of 0.25V per isolating diode, it is apparent that the solar cells must be series-connected in "strings" of 32 cells.

Hoffman normally provides cells in "shingles" so, for the Geodetic Spacecraft, eight "shingles" of four cells each can be employed. It was now necessary only to determine the number of such strings required in parallel to supply the stated intermittent and continuous loads. The total number of cells must be a multiple of 32.

3.7.2.6 Since the Geodetic Spacecraft is not deliberately oriented to the sun's rays, it is necessary to provide uniformly distributed clusters or "plaques" of solar cells. The proposed configuration comprised six such plaques. Spaced equally, these were to be located, four upon a great circle and two others at the poles thereof, so that each plaque is 90 central degrees from adjacent plaques.

3.7.2.7 Each plaque may be considered alone, as if it were the only one facing the solar radiation, since the remaining plaques will be either completely shadowed or shadowed to such extent as to render them ineffective for the purpose of charging the nickel-cadmium storage batteries. Thus, the one, effective, solar plaque must supply:

(1) A continuous load of 0.078 amperes maximum, 0.028 amperes minimum.

(2) Charging current to the nickel-cadmium storage battery system equivalent to 2.0 ampere-hours per day. Assuming a charge-discharge efficiency of 70 per cent, this corresponds to a continuous current of 0.118 amperes for a 24-hour charging period.

(3) The total requirement on the solar cells is then 0.196 amperes, maximum, 0.146 amperes minimum.

This load could be provided by either four strings (128 solar cells) or five strings (160 solar cells) of solar cells in each plaque, resulting in different operating-time capabilities as shown in table V. These calculations are based on an output per string of 0.044 amperes (i. e. 12 per cent efficient cells) less an allowance of 12 per cent for inexactness of our knowledge of the outer space solar constant and an allowance of 10 per cent for micrometeorite erosion, possible Van Allen degradation, etc. These values, accumulated over a year of operation, give a net output of 0.035 amperes per string.

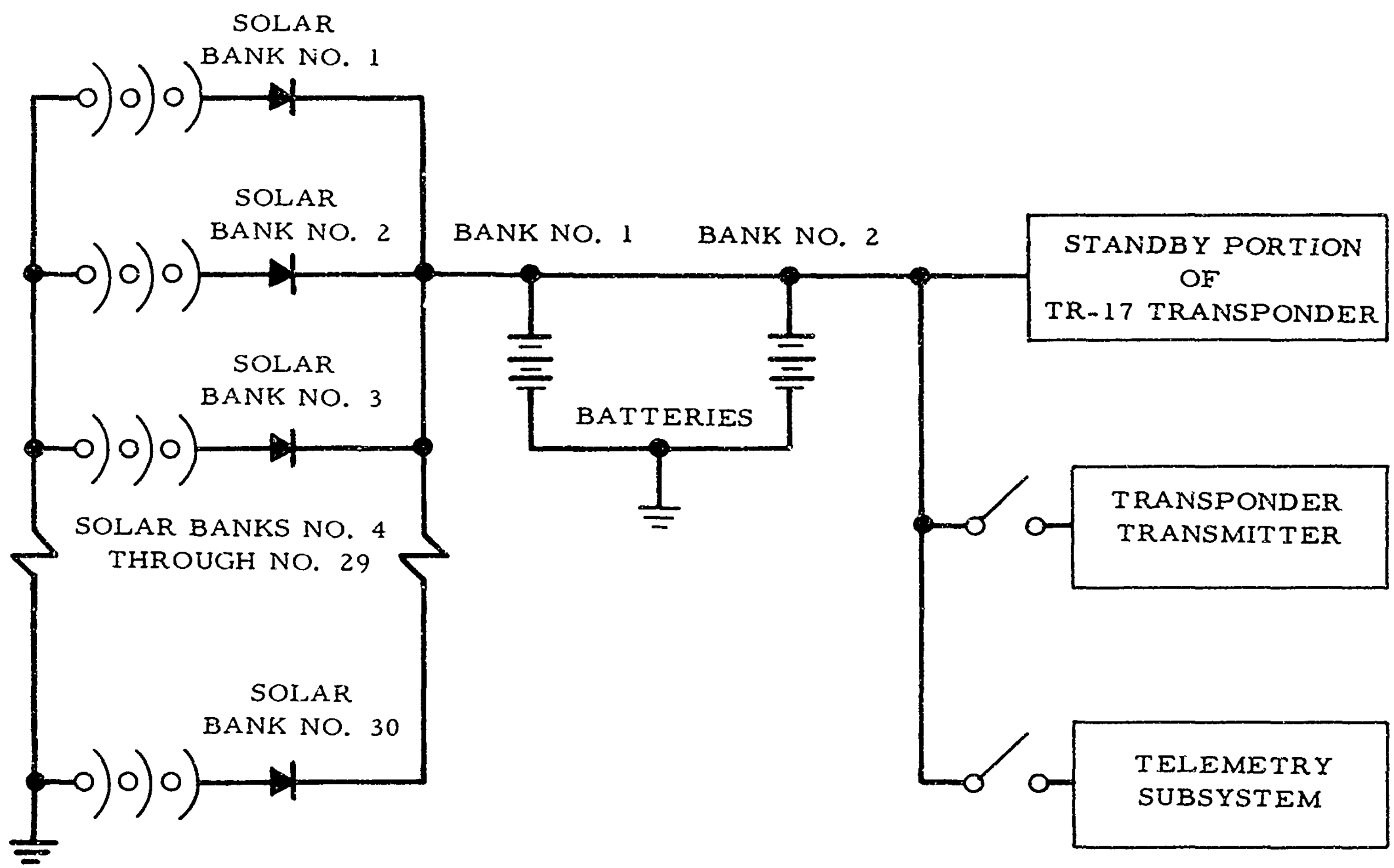


Figure 3-37. Simplified Schematic Diagram of Storage Battery/Solar Cell Connections

TABLE V

COMPARISON OF SOLAR CELL PLAQUE OPERATING TIME CAPABILITIES

Strings per plaque	Cells per plaque	Transmitter	Minutes/day 100 per cent sunlit	TR-17 operation 63 per cent sunlit
4	132	ON	31	5
4	132	OFF	56.5	30
5	160	ON	49	16
5	160	OFF	74.5	41.5

The above is calculated from the equation:

$$T = 505(I_T \cdot S - I_L) \text{ where} \tag{3.1}$$

T = time in minutes (per 24 hours) TR-17 operation

I_T = total solar-plaque current in amperes (0.140 or 0.175 amperes)

I_L = continuous load in amperes (0.078 amperes - monitor ON; 0.028 amperes - monitor OFF)

S = portion of orbit sunlit (assumed extremes of 1.00 and 0.63).

Table V shows the advantages of using five strings per plaque as selected for the Geodetic Spacecraft power supply subsystem.

3.7.3 Development Details. The following paragraphs list the finalized specifications for the power supply subsystem.

3.7.3.1 Development of the Storage Battery Package. The storage battery package consists of an aluminum battery support assembly and 20 Sonotone Type WS-103, part number 22902, cells. The cells are arranged in groups of four in a square array stacked five high. Each cell in the vertical array is separated from its adjacent cell by a teflon ring. Cells are interconnected with a short length of insulated copper wire. The battery package temperature sensor is inserted into the array, and the entire assembly potted in Eccosil 4640. (See figure 3-38.) A circular chassis fitted to the top of the package contains the transponder power converter. (See figure 3-39.) All electrical leads are routed through the top plate of the power converter chassis. Specifications are as follows:

(1) Over-all length - 12.312 ± 0.016 inches

(2) Over-all diameter - 3.230 inches

(3) Weight - 6.7 pounds

(4) Charge current - 300 ma constant at 16 hours

3.7.3.2 Development of the Solar Cell Plaques. Each solar cell plaque consists of 160 solar cells arranged in five banks of 32 cells each. (See figure 3-40.) Each cell is equipped with an optical filter and a coating to provide thermal stabilization. The cell banks are mounted on a honeycombed plate which contains the interconnecting wiring and the isolation diodes. (See figures 3-41 and 3-42.) The exposed area of the outer panel is painted with white reflective paint (Trial Chemical #293324 or equivalent). Specifications are as follows:

(1) Outer diameter of honeycombed plate - 8.678 $^{+0.000}_{-0.015}$ inches

(2) Type of honeycomb - Aluminum alloy 5052-H39 1/8-inch hex X, 0.0007P X :375

(3) Total thickness - 0.415 inch (including top and bottom caps, excluding solar cells)

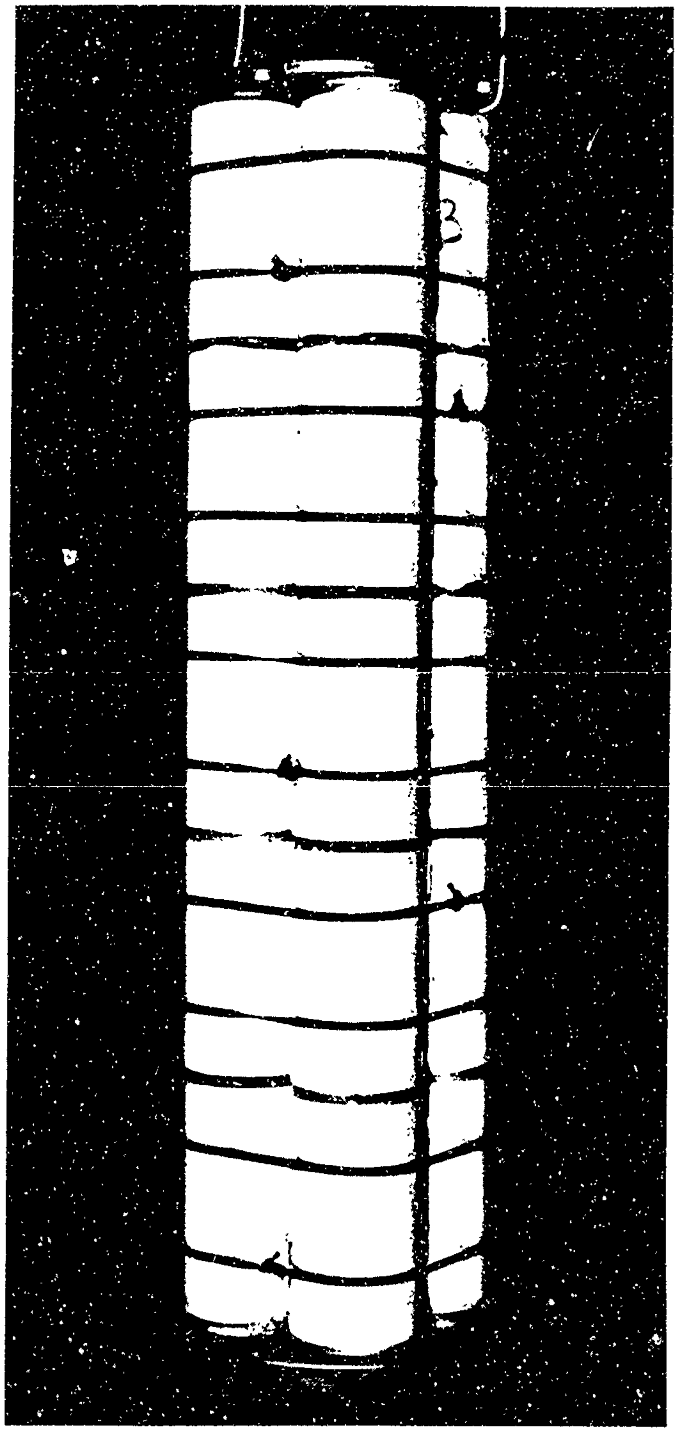


Figure 3-38. Battery Package Before Potting

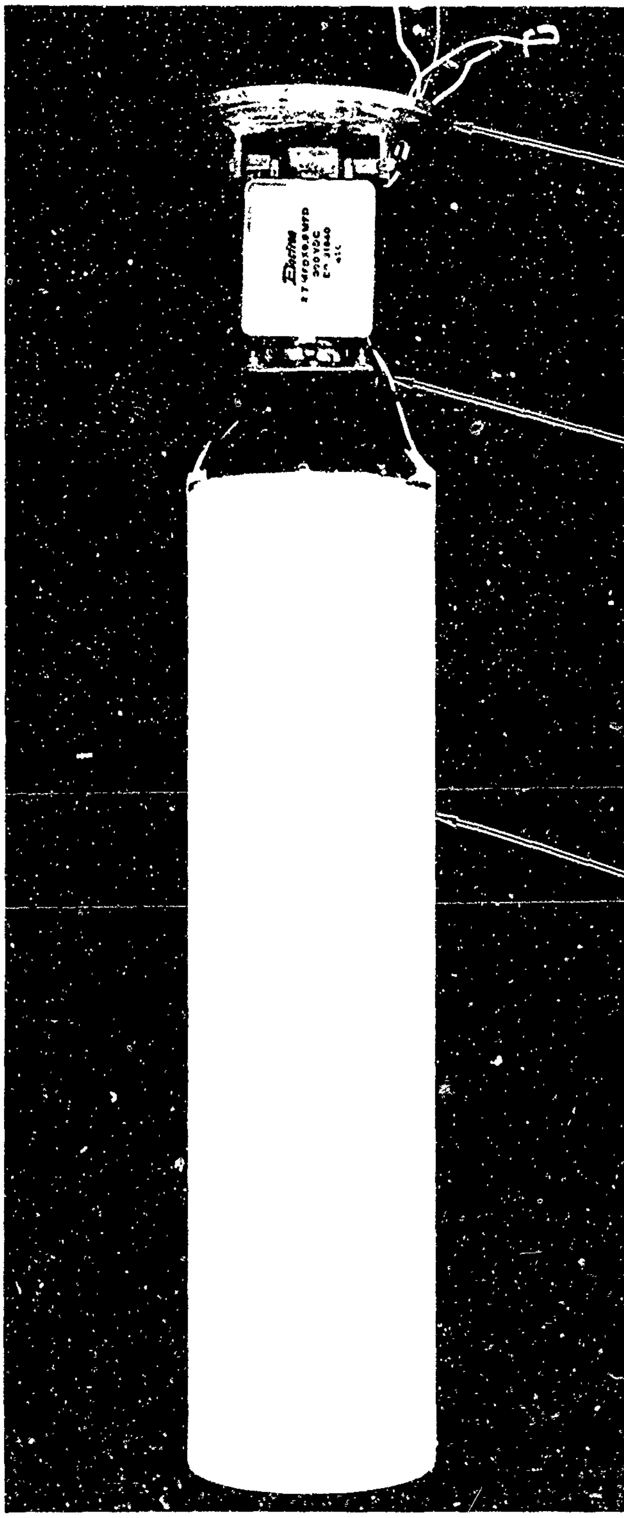


Figure 3-39. Potted Battery Package with a Power Converter Attached

(4) Top cap - 0.010 inch thick aluminum alloy 20424-T4

(5) Bottom cap - 0.032 inch thick aluminum alloy 6061.T x $9.065\,^{+0.000}_{-0.010}$ inch outer diameter, with exposed surface flat to 0.030 inch

(6) Over-all efficiency of each solar cell shingle - 12 per cent as measured under standard test conditions (normal incidence illumination from tungsten light source with a color temperature of 2800° K, at a light level which corresponds to 100 mw/sq cm as correlated by means of standard solar cell); cell temperature at test is $26 \pm 2^{\circ}$ C, unfiltered cells

(7) Solar cell type - Hoffman Electronics Corporation type 120CG or equivalent

(8) Individual cell output - 0.485V under proper exposure

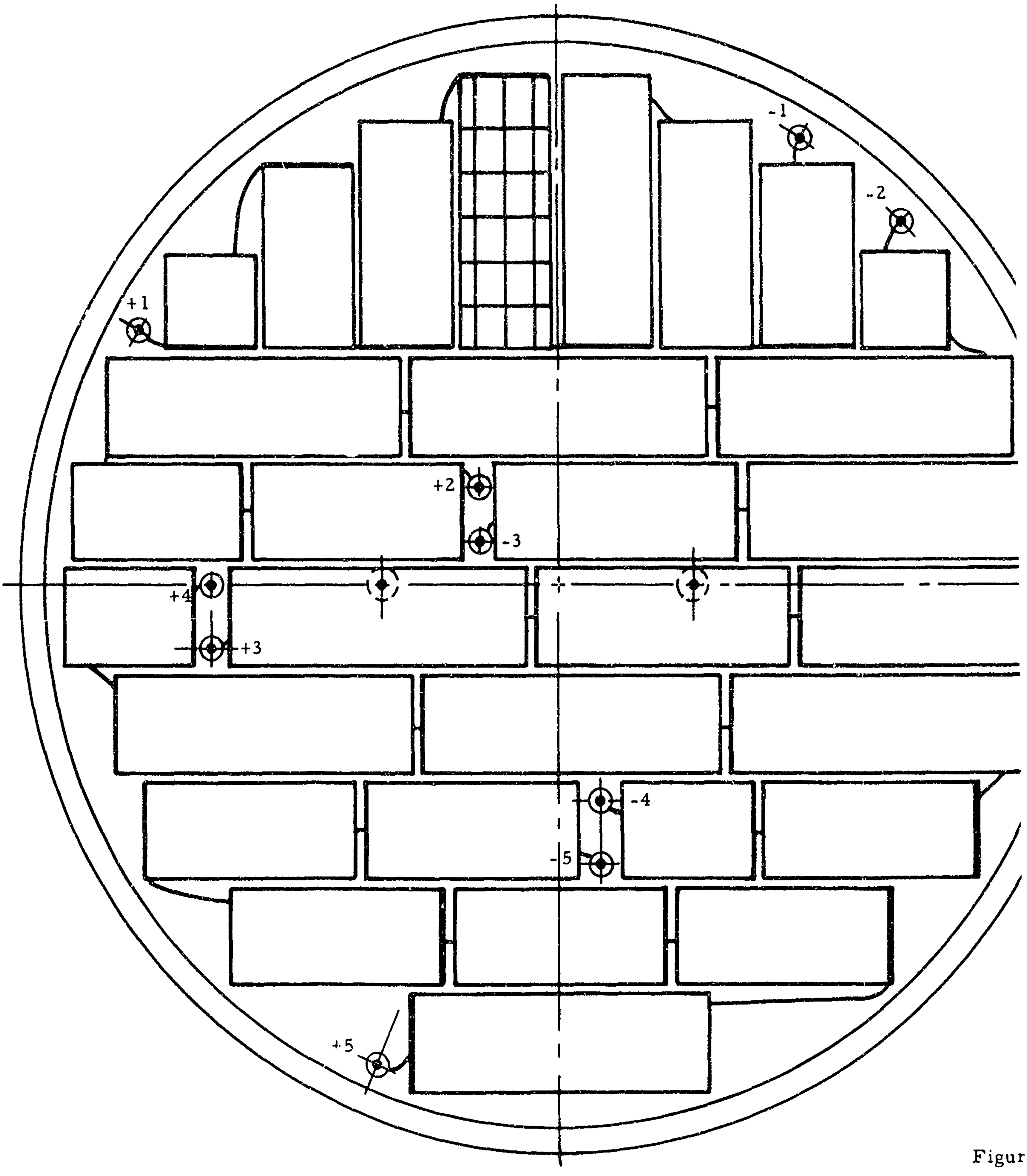
(9) Cell bank output - 15.5V per band open circuitry after installation of filter

(10) Output of solar cell string (32 cells) - At least 40 ma at 12.5V under standard test conditions as in (6) above (assembled with filters and diodes)

(11) Type of optical filter - Ocli type 207Scc450-1, with UV and AR coatings

(12) Transmission characteristics of Ocli filter

Wavelength (mu)	Transmission
300-400	Less than 1 per cent average
450 ±15	50 per cent
500-1000	90 per cent or greater average



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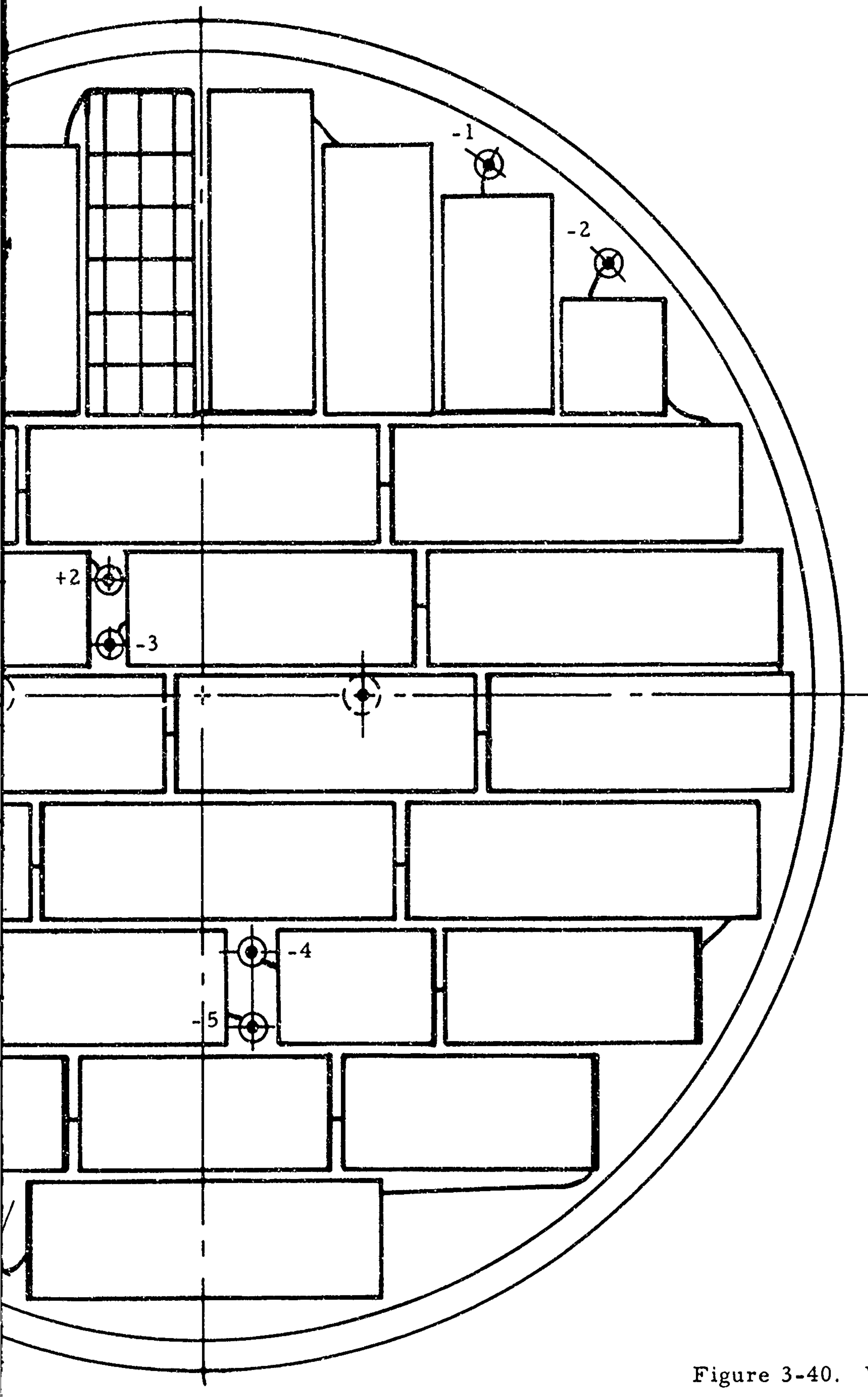


Figure 3-40. Wiring Diagram of Typical Solar Cell Plaque

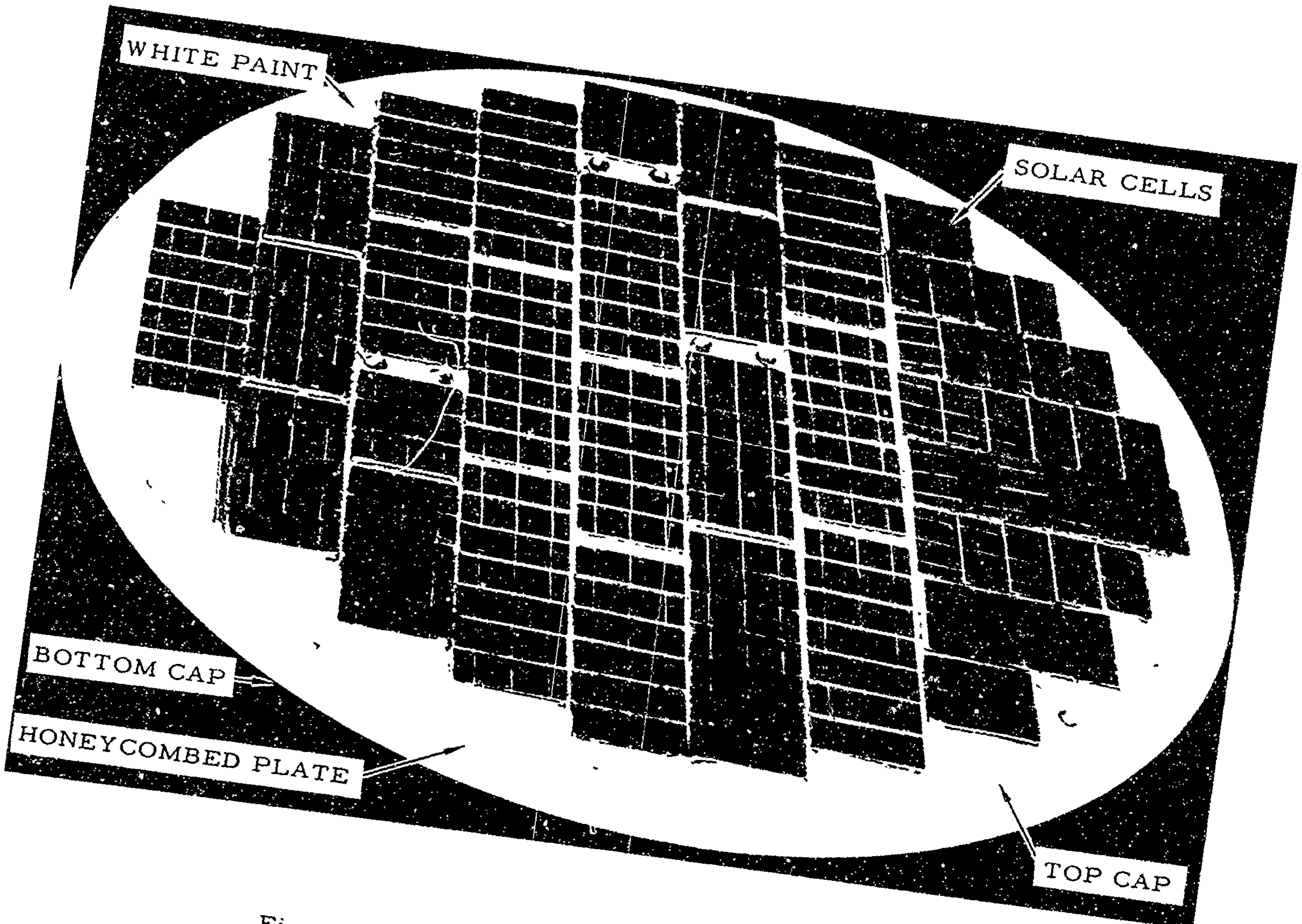


Figure 3-41. Typical Solar Cell Plaque (Top View)

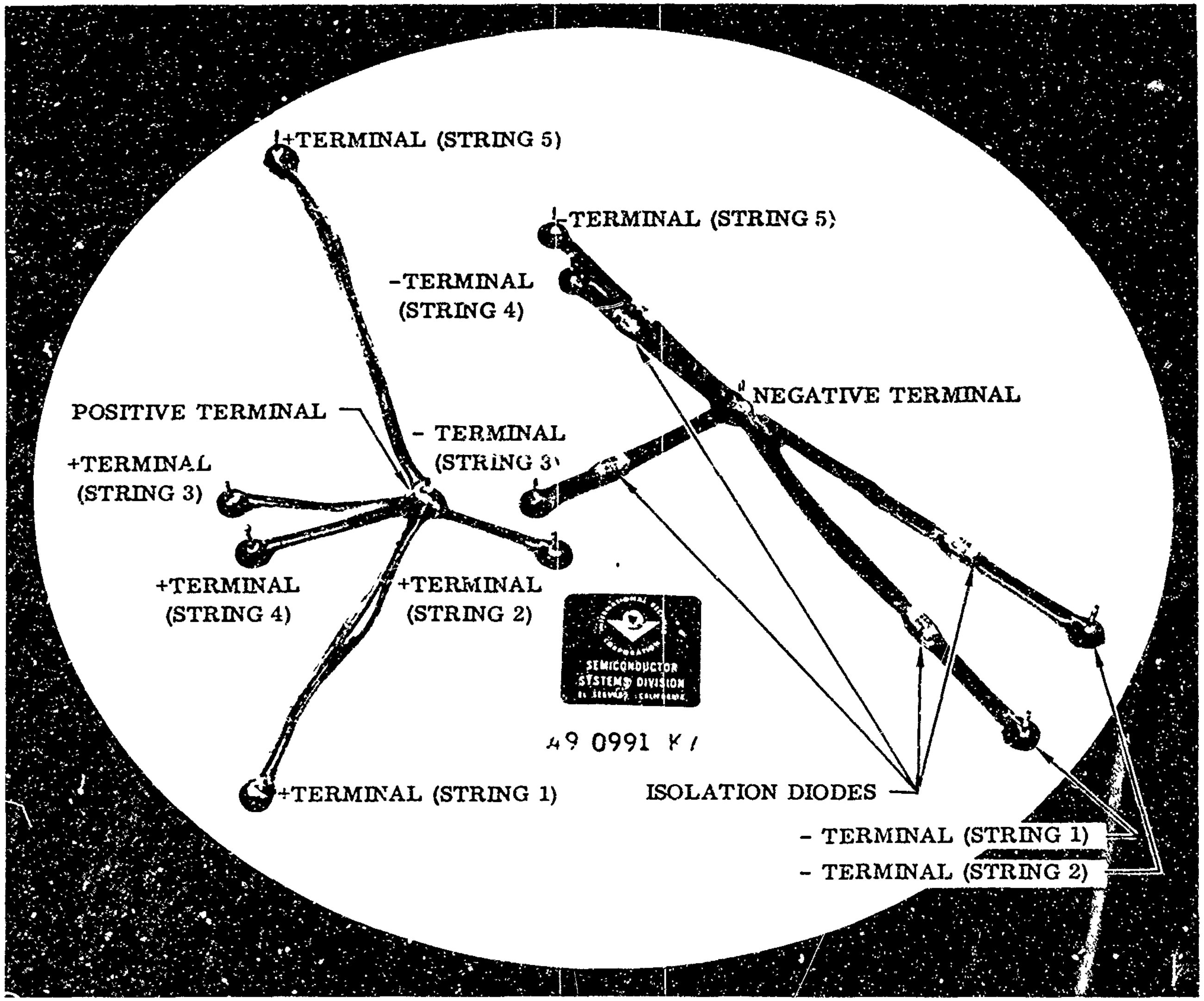


Figure 3-42. Typical Solar Cell Plaque (Bottom View)

(13) α/ϵ for silicon solar cell and filter cover combination -

$\alpha = 0.78$
$\epsilon = 0.82$ at $250^{\circ}K$
$\epsilon = 0.83$ at $300^{\circ}K$

(14) α/ϵ of Trial Chemical paint -

$\alpha = 0.32$
$\epsilon = 0.92$

3.7.4 Investigation for Qualified Manufacturers. The basic configuration and characteristics of the standard storage cell conformed specifically to the spacecraft requirements. The nickel-cadmium storage batteries were purchased from the Sonotone Corporation who is the only manufacturer of a spacecraft-proven battery. With regard to the solar cell plaques, Cubic contacted Hoffman Electronics Corporation, International Rectifier Corporation, and Solar Systems Inc. Hoffman fabricated the six plaques for the prototype spacecraft, and International Rectifier constructed the units used on the flight models.

3.7.5 Fabrication of the Battery Package. Cubic constructed three identical battery packages. Initially the battery cells were separated vertically by a slotted teflon ring supporting a beryllium-copper strip which was soldered to the battery terminals. However, the soldered joint would not resist twisting stress, and this configuration was rejected. The Teflon cushions were retained, and a wire was soldered to the battery pairs. The improved connection also facilitated assembly, as the length of wire could be rotated around the teflon ring to take up its slack. During the potting procedures, care is taken to avoid filling the central tube in the battery support assembly. The tube serves as an air vent and is designed into the assembly to facilitate insertion of the assembly into the instrumentation tube.

3.7.6 Fabrication of the Solar Cell Plaques. Fabrication and delivery of the solar cell plaques formed a limiting factor in the final delivery of the Spacecraft. Although the purchase order was placed with Hoffman on 7 March for the first units, it was 7 April before

all formal negotiations were complete; Hoffman entered the order with a desired shipment date of three weeks but not more than five weeks. On 21 April Hoffman indicated a 16 May delivery; investigation showed that they had just obtained the honeycomb panels and were beginning to assemble the cells. The six solar cell plaques for the prototype spacecraft were finally delivered on 16 May 1961.

3.7.7 International Rectifier Corporation also experienced some delays in constructing the 12 solar cell plaques for the spacecraft flight models. On 3 July, they informed Cubic that, of the 14 honeycombed panels ordered for the project, eight were delivered to them undersize. On 6 July 1961 Cubic instructed them to accept six of the undersize panels and utilize them for the first set of plaques. On 6 July International Rectifier informed Cubic that they were having trouble with short circuits in the first set of plaques. The difficulty resulted from the absence of a hard anodize which is not permitted by specification. However, the difficulties were resolved and the first set of plaques was delivered on 15 July, approximately two weeks late. The delivery of the second set of plaques was completed by 24 July.

3.7.8 Testing of the Battery Package. In order to verify minimum performance standards of the three battery packages, Cubic performed three charge and discharge cycles for each package. Table VI is a list of the test equipment used; figure 3-43 shows the test set-up. The packages were charged using 12V at 300 ma for 16 hours, then discharged at a minimum of 10.7V and 1.9 amperes. All packages completed the cycling within the specified performance standards.

3.7.9 Testing of the Solar Cell Plaques. All solar cell plaques were tested and certified by the manufacturers prior to delivery. Cubic performed open-circuit voltage and closed-circuit current checks on each plaque prior to installation in the Spacecraft. Table VII lists the test equipment utilized for the check. Figure 3-44 shows the test setup. The over-all efficiency of each plaque was verified to be within the required 12 per cent. Refer to table VIII for open-circuit voltage and closed-circuit current readings of a typical solar cell plaque.

3.8 Developmental History of the Antenna Subsystem. Cubic designed, fabricated, and installed the antennas in the three Geodetic Spacecraft. Temec, Inc. performed antenna pattern tests to confirm

TABLE VI

TEST EQUIPMENT UTILIZED
FOR
BATTERY PACKAGE CHECKOUT

Description	Manufacturer	Model No.
DC voltage source	Kepco	800
Volt-ohmmeter	Simpson	260
Power resistor	Clarostat	240C
Ammeter	Simpson	260

TABLE VII

TEST EQUIPMENT UTILIZED
FOR
SOLAR CELL PLAQUE CHECKOUT

Description	Manufacturer	Model No.
Ammeter	Simpson	260
Voltmeter	Simpson	260
Power Resistor	Clarostat	240C

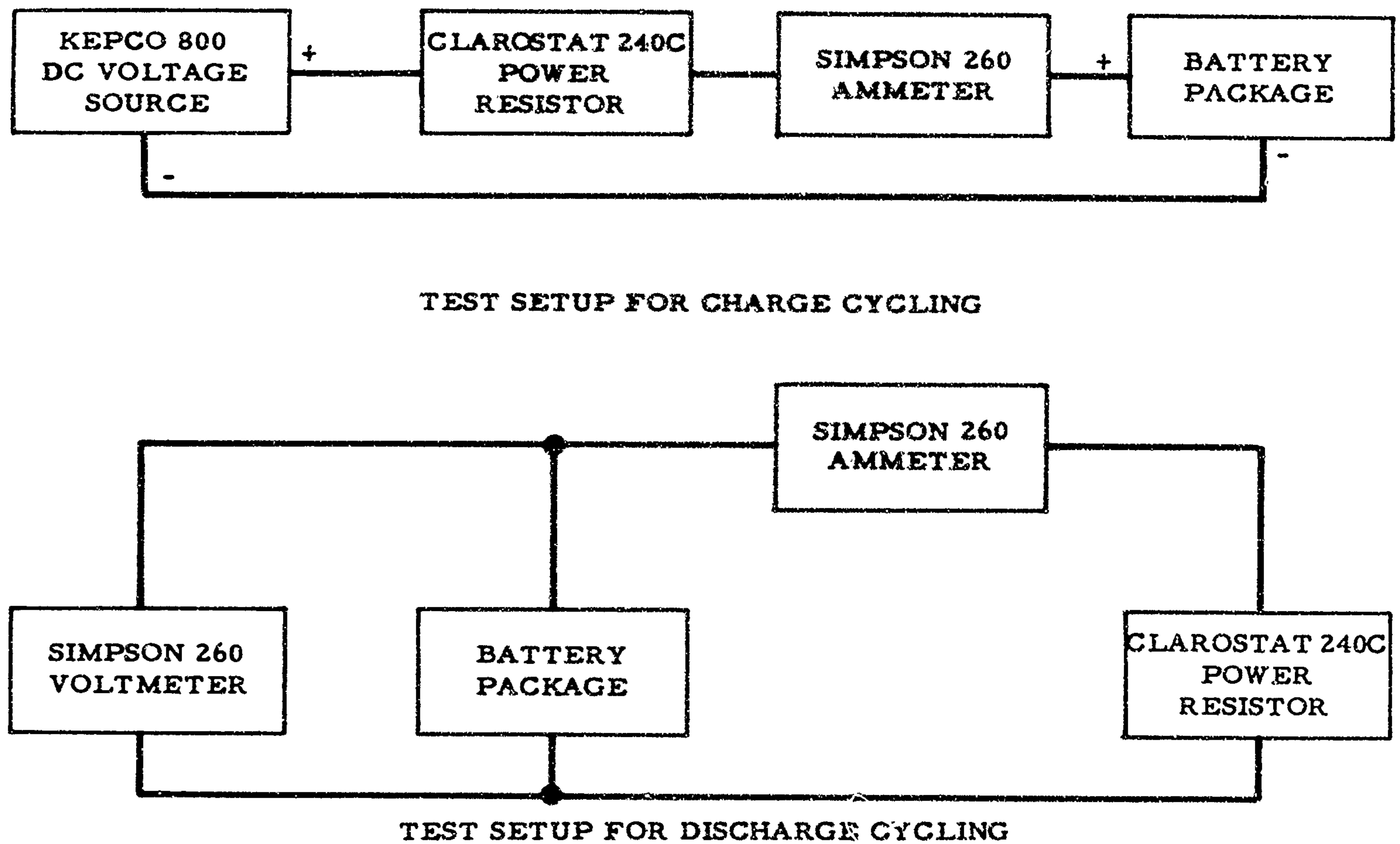


Figure 3-43. Test Setup for Battery Package Checkout

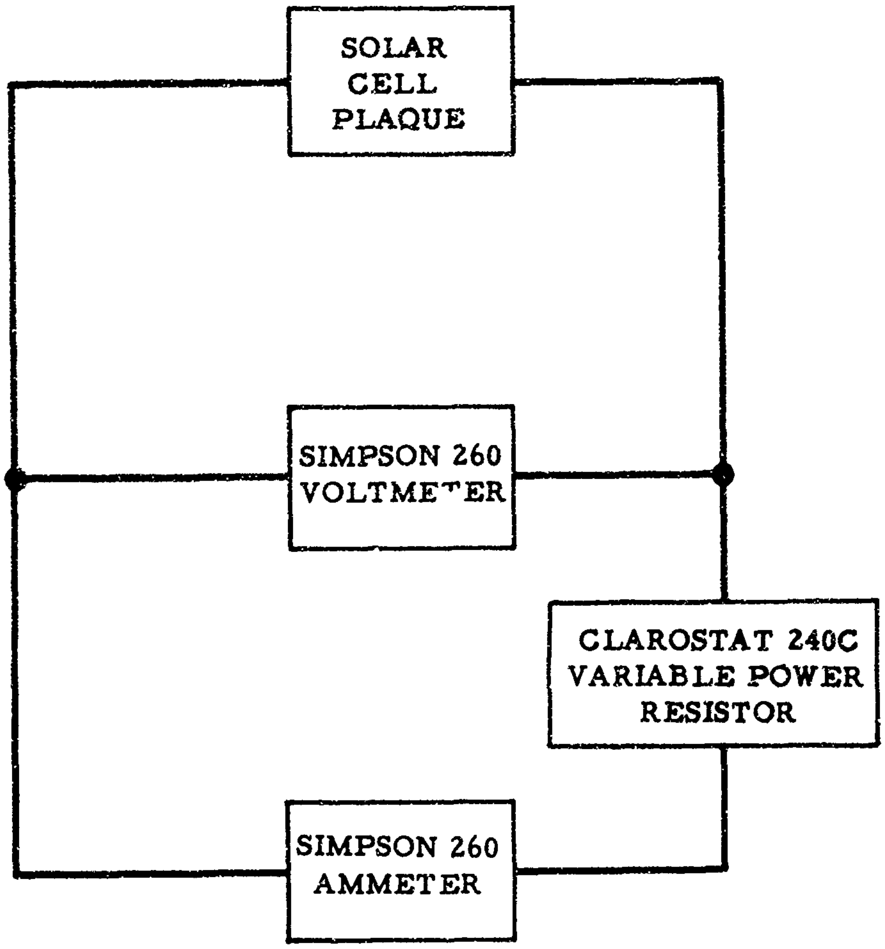


Figure 3-44. Test Setup for Solar Cell Plaque Checkout

TABLE VIII

OPEN-CIRCUIT VOLTAGE AND CLOSED-CIRCUIT CURRENT
FOR
TYPICAL SOLAR CELL PLAQUE

Date test performed: 26 July 1961

Time: 10:30 AM, PDT

Conditions: Bright sunlight, slight horizon haze

String No.	Open-Circuit Volts	Short-Circuit MA	Current (MA at 12V Loading)	Resistance (Ohms for 12V Loading)
1	16.3	42.0	40.5	286
2	16.3	40.0	38.0	307
3	16.4	41.0	39.0	298
4	16.4	40.5	38.5	300
5	16.5	40.5	39.2	295
Measured Total	16.5	205	190	61

compatibility with the Spacecraft. Figure 3-45 shows the location of the antennas with respect to the Spacecraft.

3.8.1 General Problem. The general problem in the development of the antenna subsystem involved the design of an omnidirectional radiator for the 136.11 mc, 224.5 mc, and 421-449 mc spacecraft frequencies.

3.8.2 Preliminary Considerations. The Geodetic Spacecraft required an integrated/omnidirectional antenna subsystem to handle the TR-17 transponder and telemetry signals received and radiated while in orbit. The antennas receive ground-initiated transponder signals at 421 mc and re-radiate these signals at 224.5 mc and 449 mc. A telemetry antenna must radiate signals at 136.11 mc without creating interference with the SECOR system. Consideration was given to the radiation characteristics of the sphere, and its specific effect on a proposed antenna subsystem. Moreover, the individual antennas had to be mechanically designed to permit folding and erection.

3.8.3 Cubic designed the antenna subsystem and determined the number, type, and position of the elements. Temec, Inc. performed tests to verify the radiation patterns and determine the optimum method of impedance matching. In the original configuration, the 136.11 mc antenna system was a two-element array of monopoles, located on the north and south poles of the satellite. However, this configuration was replaced by a single monopole at the north pole of the Spacecraft when radiation patterns demonstrated that the additional antenna did not appreciably increase radiation. During the pattern tests attempts were made to use common antennas for the 136.11 mc and 224.5 mc frequencies, or the 224.5 mc and 449 mc frequencies. The following paragraphs describe the test performed by Temec, Inc. Table IX is a list of the test equipment used. Each piece of equipment used had been calibrated within two months prior to use for this project. Calibration records and recall dates for this equipment are available for inspection at Temec, Inc.

3.8.3.1 Radiation Pattern Tests for the 136.11 MC Antenna. The 136 mc antenna system is a one-quarter wavelength ~~dipole~~ monopole located on the north pole of the satellite. Principal plane radiation patterns were recorded for both vertical (E_θ) and horizontal (E_ϕ) polarization on a one-third scale model of the satellite, at a scale frequency of 408 mc. The coordinate system used in recording these

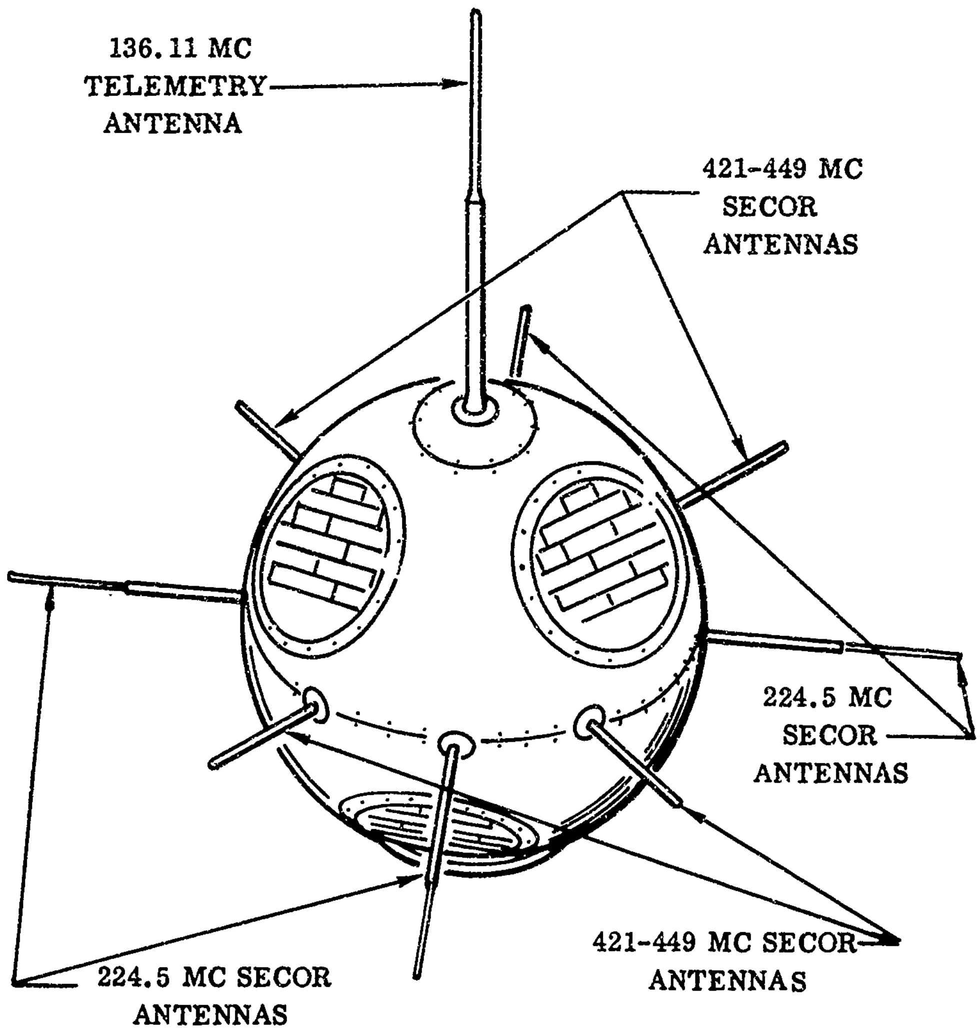


Figure 3-45. Location of the Geodetic Spacecraft Antennas

TABLE IX

TEST EQUIPMENT UTILIZED FOR IMPEDANCE AND RADIATION MEASUREMENTS OF ANTENNA SUBSYSTEM

Description	Manufacturer	Model No.
IMPEDANCE		
Standing-wave detector	Polytechnic Research and Development Co.	219
Square-wave generator	Hewlett Packard	211A
VHF signal generator	Hewlett Packard	608C
Standing-wave detector	Hewlett Packard	415B
RADIATION		
Polar recorder	Antlab	
Selective filter amplifier	Antlab	
Recorder servo amplifier	Antlab	
Test oscillator	FXR	L772A
Tuning-fork modulator	Antlab	
VHF signal generator	Hewlett Packard	608C
Broadband detector	FXR	N210B

radiation patterns is shown in figure 3-46. The recorded radiation patterns are shown in figure 3-47. An examination of the radiation patterns clearly shows that the antenna system is vertically polarized, with minima present in the direction of the north pole. The antenna system is omnidirectional and will provide radiation coverage of approximately 60 degrees in elevation.

3.8.3.2 Radiation Pattern Tests for the 224.5 MC Antenna Array. The 224.5 mc antenna system is a 4-element array of quarter-wavelength monopole antennas, spaced 90 degrees apart around the equator of the satellite. Each element is fed 90 degrees out of phase with respect to its adjacent element in a turnstile manner, to provide both vertical and horizontal radiation coverage. (See figure 3-48.) Principal plane patterns were recorded for both vertical and horizontal polarization on a one-fifth scale model of the satellite at a scale frequency of 1125 mc. The recorded radiation patterns are shown in figure 3-49. An examination of the radiation patterns indicates that the antenna system is both vertically and horizontally polarized, with horizontal polarization being dominant, as would be expected for the array orientation. A comparison of the vertical and horizontal radiation pattern for any given plane (eg., E_θ, $\theta = 0$; and E_ϕ, $\theta = 0$) shows that coverage is maintained for all aspects of satellite orientation. The deepest minima observed is -16 db for the E_ϕ, $\theta = 0$ cut.

3.8.3.3 Radiation Pattern Tests for 421-449 MC Antenna Array. The 421-449 mc antenna system is a 4-element array of monopole antennas spaced 90 degrees apart, and 45 degrees from the 224.5 mc antenna system, around the equator of the satellite. The monopole antennas are each one-quarter wavelength in length (f_0 = 437 mc). The elements are fed like the 224.5 mc antenna system. (See figure 3-50.) Principal plane radiation patterns were recorded for both vertical and horizontal polarization on a one-third scale model of the satellite at scale frequencies of 1263 mc and 1347 mc. An examination of the radiation patterns corresponding to a frequency of 421 mc shows that the deepest minima is approximately -31 db for both E_θ and E_ϕ, $\phi = 0$, $\theta = 60^\circ$. This minima also occurs at 449 mc. The radiation patterns are shown in figure 3-51 for the 421 mc frequency, and in figure 3-52 for the 440 mc frequency.

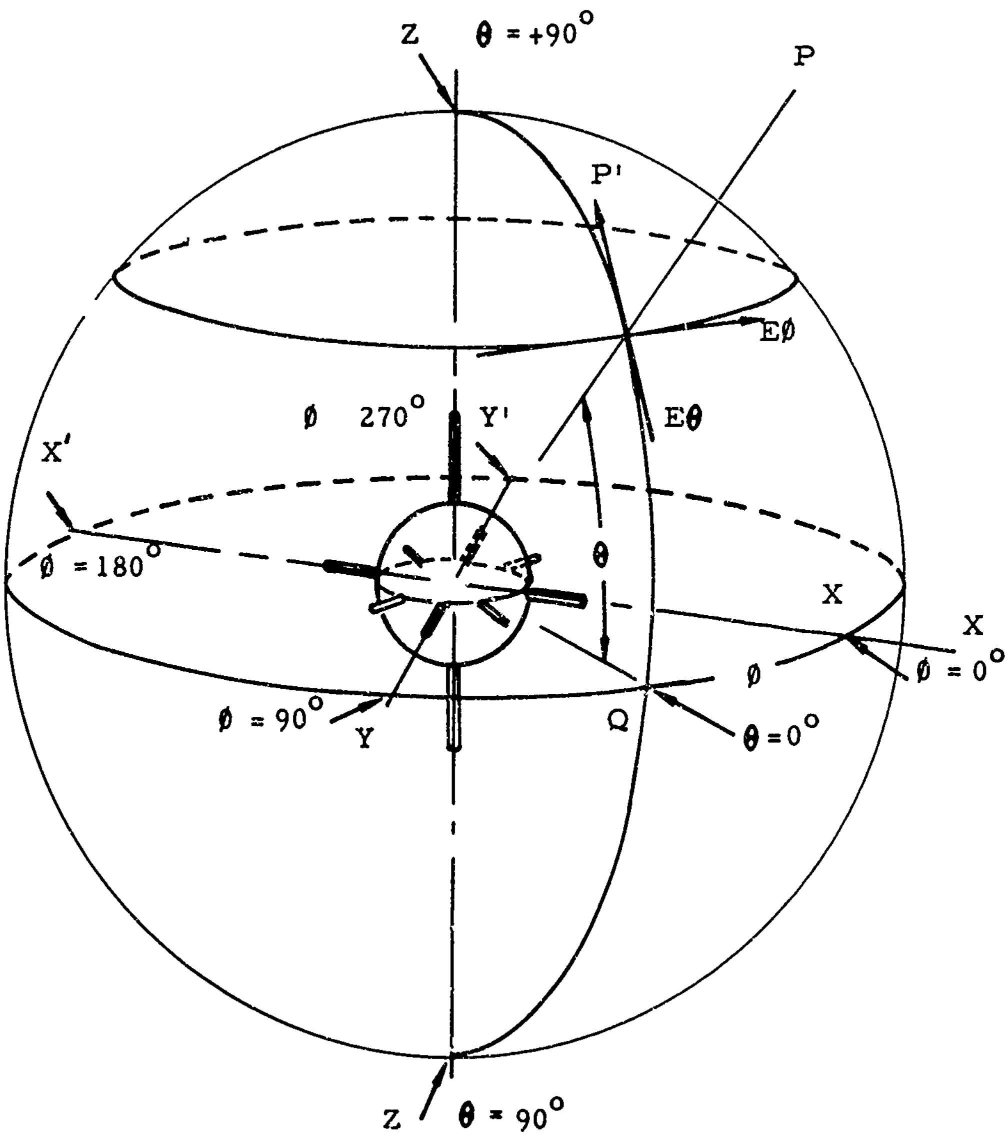


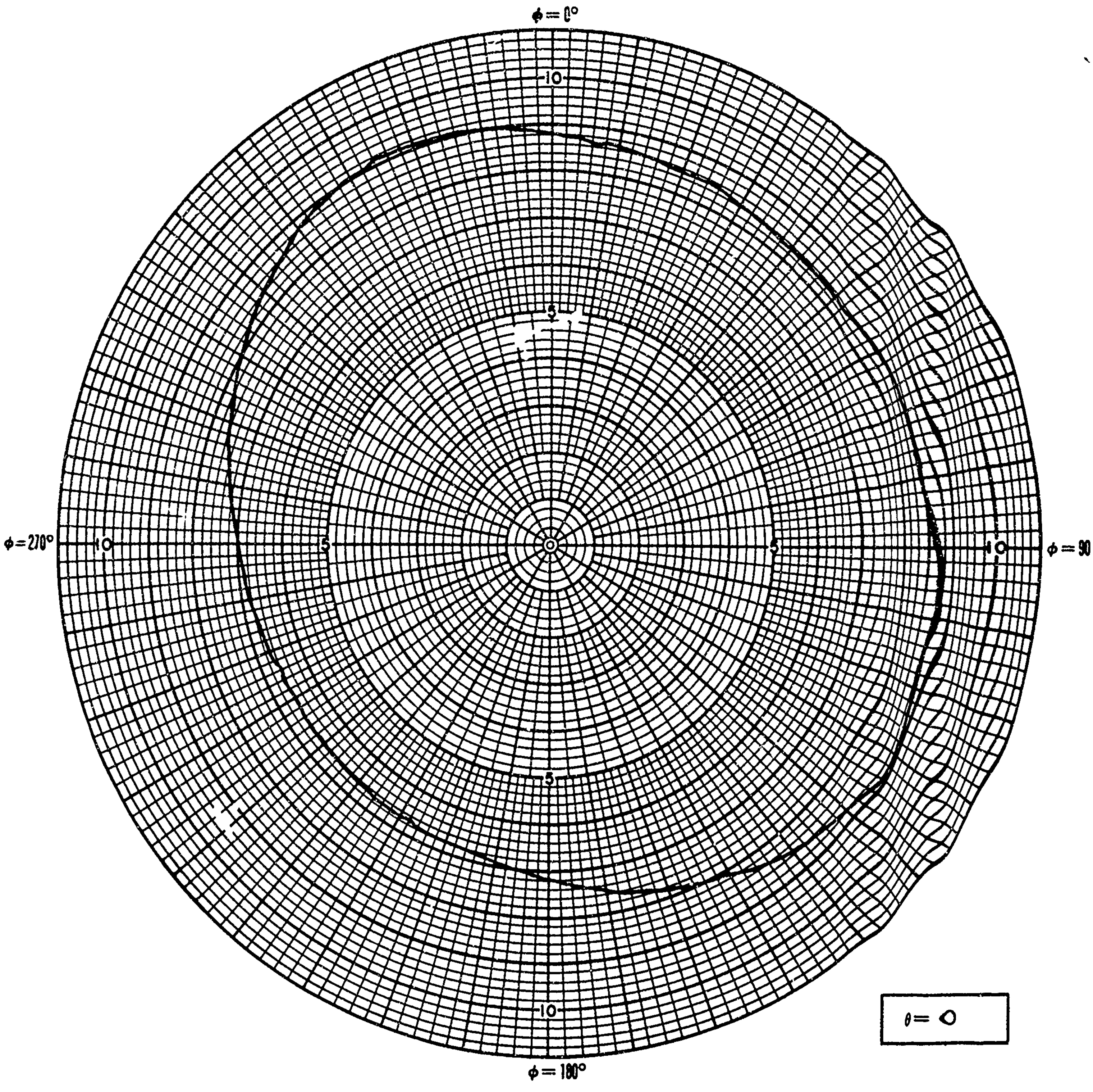
Figure 3-46. Radiation Pattern Coordinate System

3.8.3.4 Impedance Measurements. Impedance measurements were made on a full scale mockup of the Spacecraft. For each antenna system, each element was measured individually to determine the optimum method of impedance-matching. For all three antenna systems it was decided to use short-circuited parallel stubs at the input to the array.

3.8.4 Development Details. The final antenna system design provides omnidirectional coverage as nearly as possible for all frequencies. The dipole was chosen as the basic antenna element for all frequencies. This choice was dictated by the physical factors involved such as the number of antennas required, and the size of the Spacecraft. Attempts were made to use common antennas for 136 mc and 224.5 mc frequencies, or the 224.5 mc and 449 mc frequencies. However, because of the wide separation of frequencies and difficulty in matching this proved to be impracticable. In the final design the Geodetic Spacecraft uses three separate and independent antenna systems; the 136 mc telemetry transmitting antenna, the 224.5 mc SECOR one-half frequency transmitting antenna array, and 420-450 mc SECOR antenna array. This latter antenna system provides for receiving at 421 mc and transmitting at 449 mc.

3.8.4.1 Development of 136.11 MC Antenna. The least significant of the antenna systems is the 136 mc telemetry transmitter. This system is used to provide information on the performance of the vehicle while in orbit. It also provides a signal for initial tracking and initial orbit computation. It is not directly involved in the primary mission of the Geodetic Spacecraft. The 136 mc antenna system is a single 1/4-wavelength ~~dipole~~ Monopole attached at the north pole of the Spacecraft. The antenna, made of aluminum tubing, is spring-loaded and hinged in the middle to allow folding to fit inside the SCOUT vehicle nose faring. Impedance measurements required the removal of 0.870 inch from the original length of the antenna in order to eliminate error. The antenna pattern is the typical doughnut shown pictorially in figure 3-53.

3.8.4.2 Development of 224.5 MC Antenna Array. The 224.5 mc antenna system is used in conjunction with the 1 watt 1/2-frequency transmitter of the SECOR transponder. This signal is modulated by the highest ranging time and used in the ionospheric correction process. This antenna system is similar,



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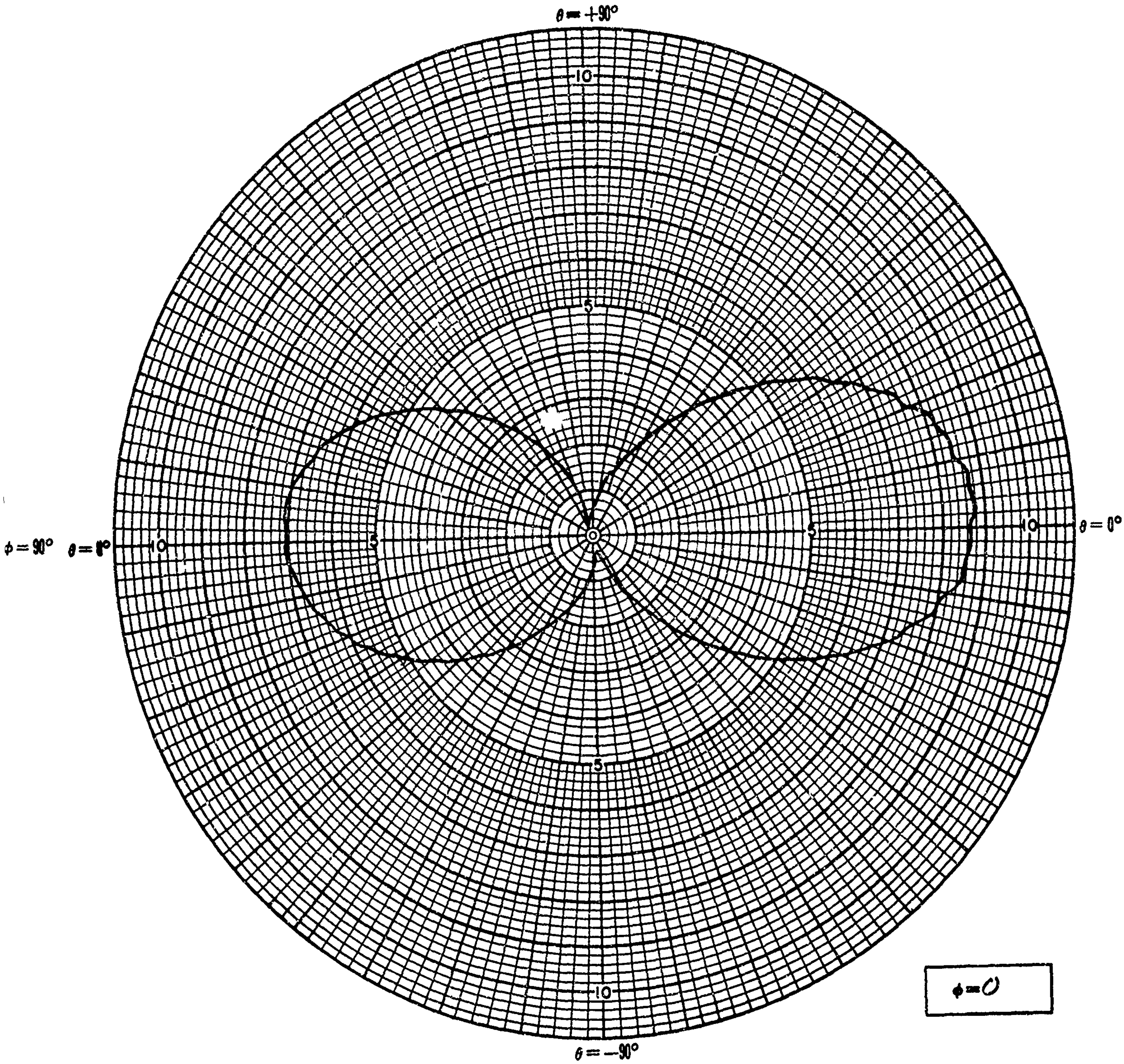
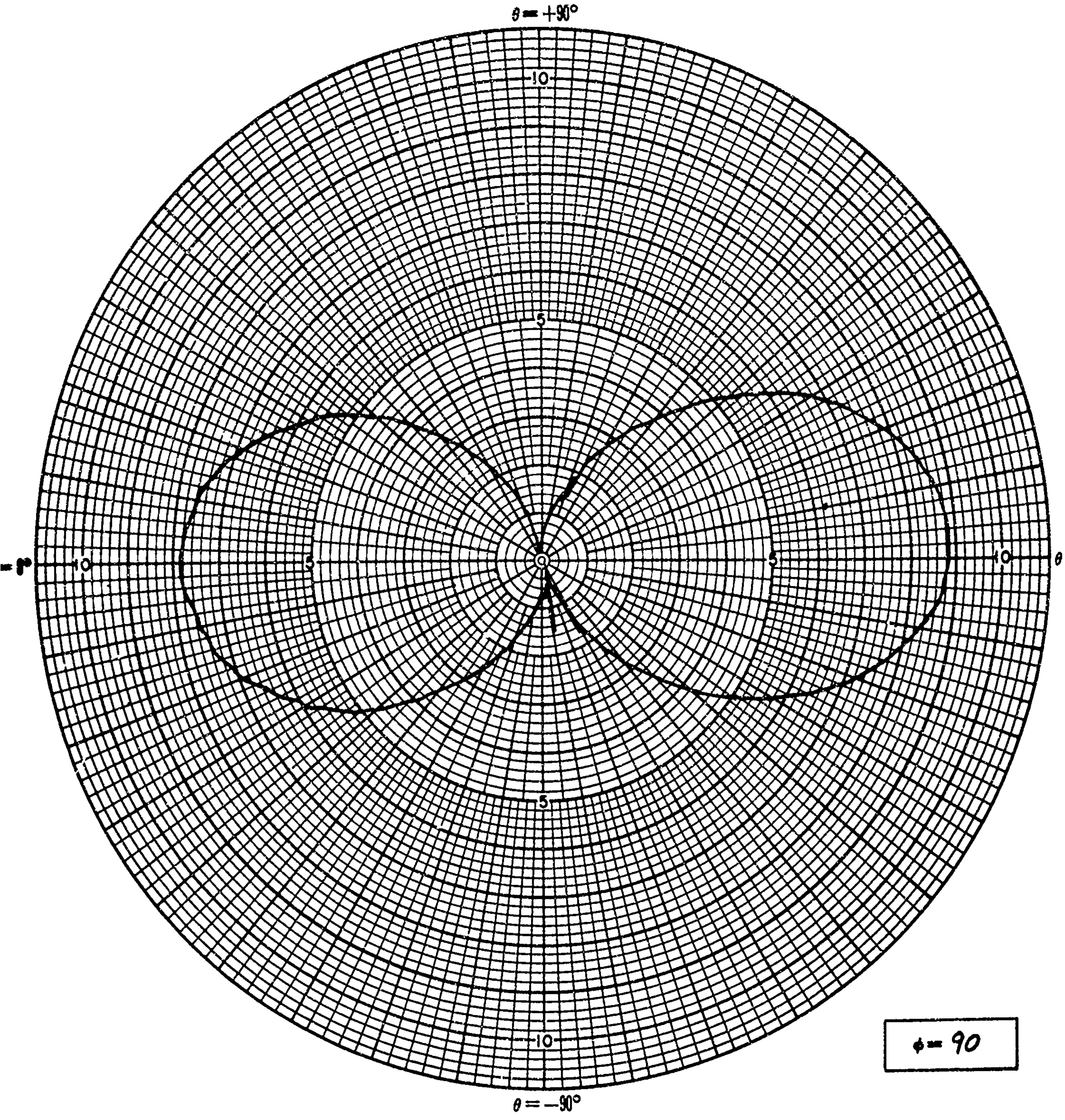


Figure 3-47. 136.11 MC Antenna
Radiation Patterns (Sheet 1 of 3)

B



φ = 270°

A

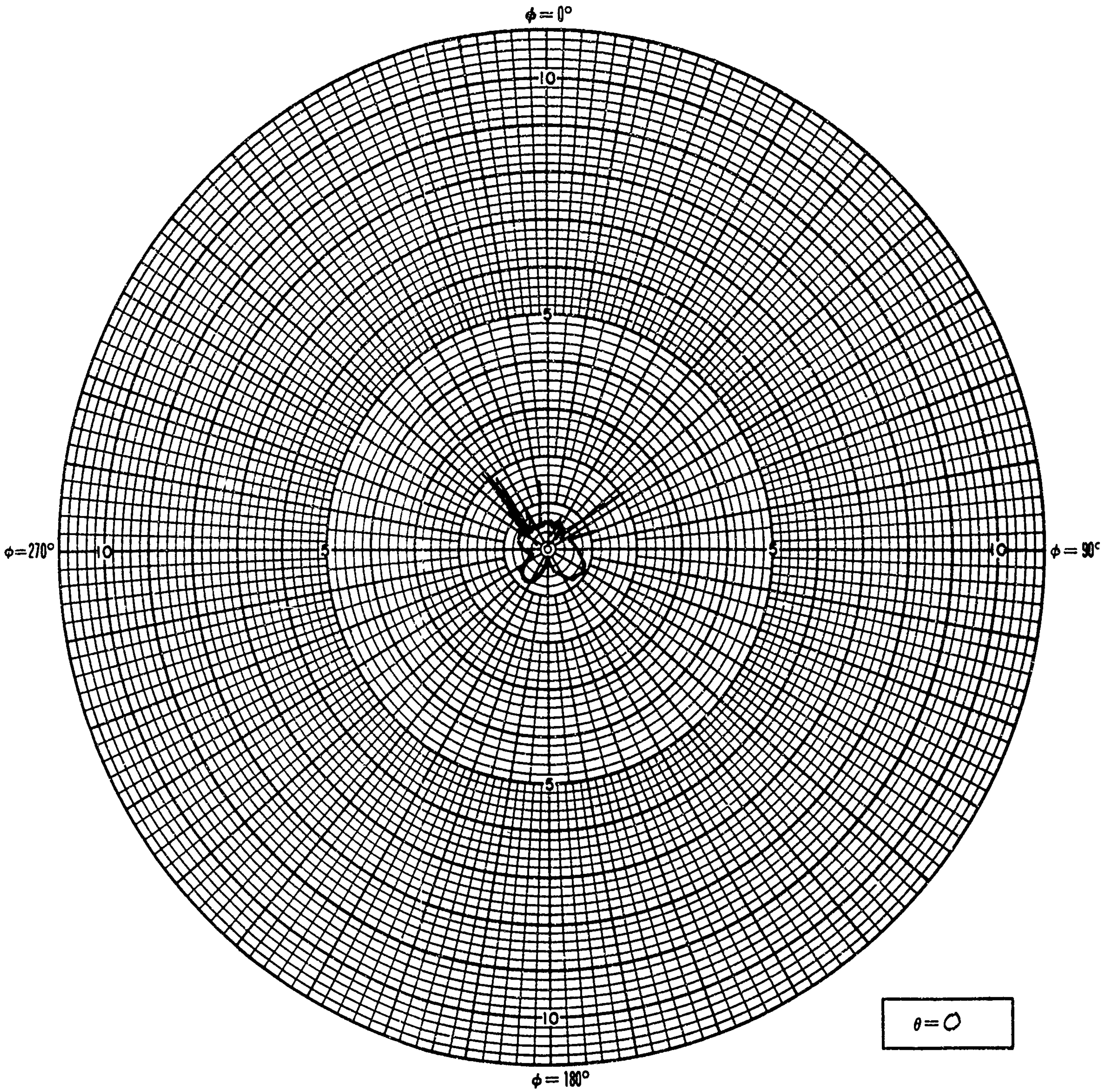
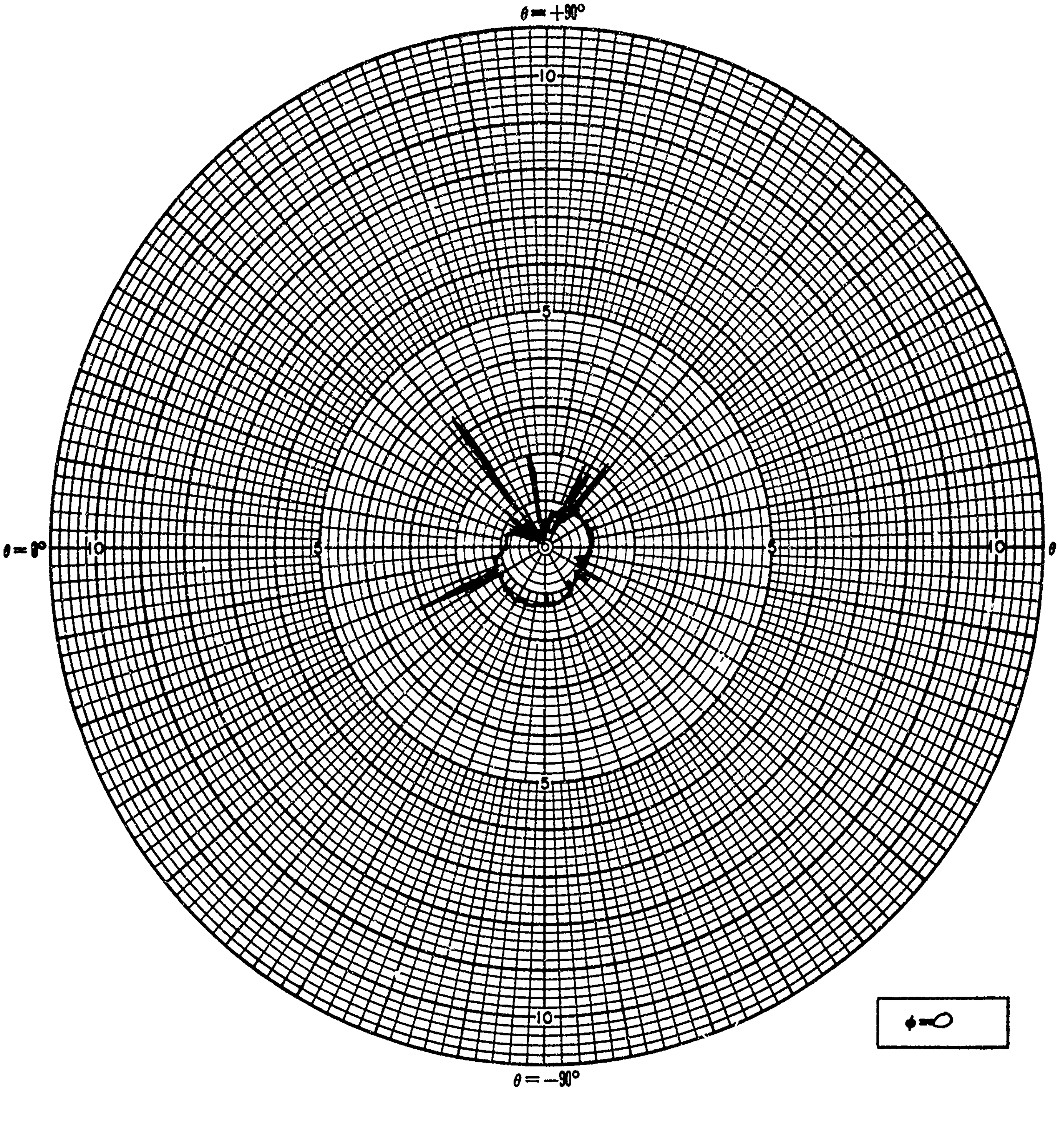


Figure 3-47. 136.11 MC Antenna Radiation Patterns (Sheet 2 of 3)

B



θ = 0°

A

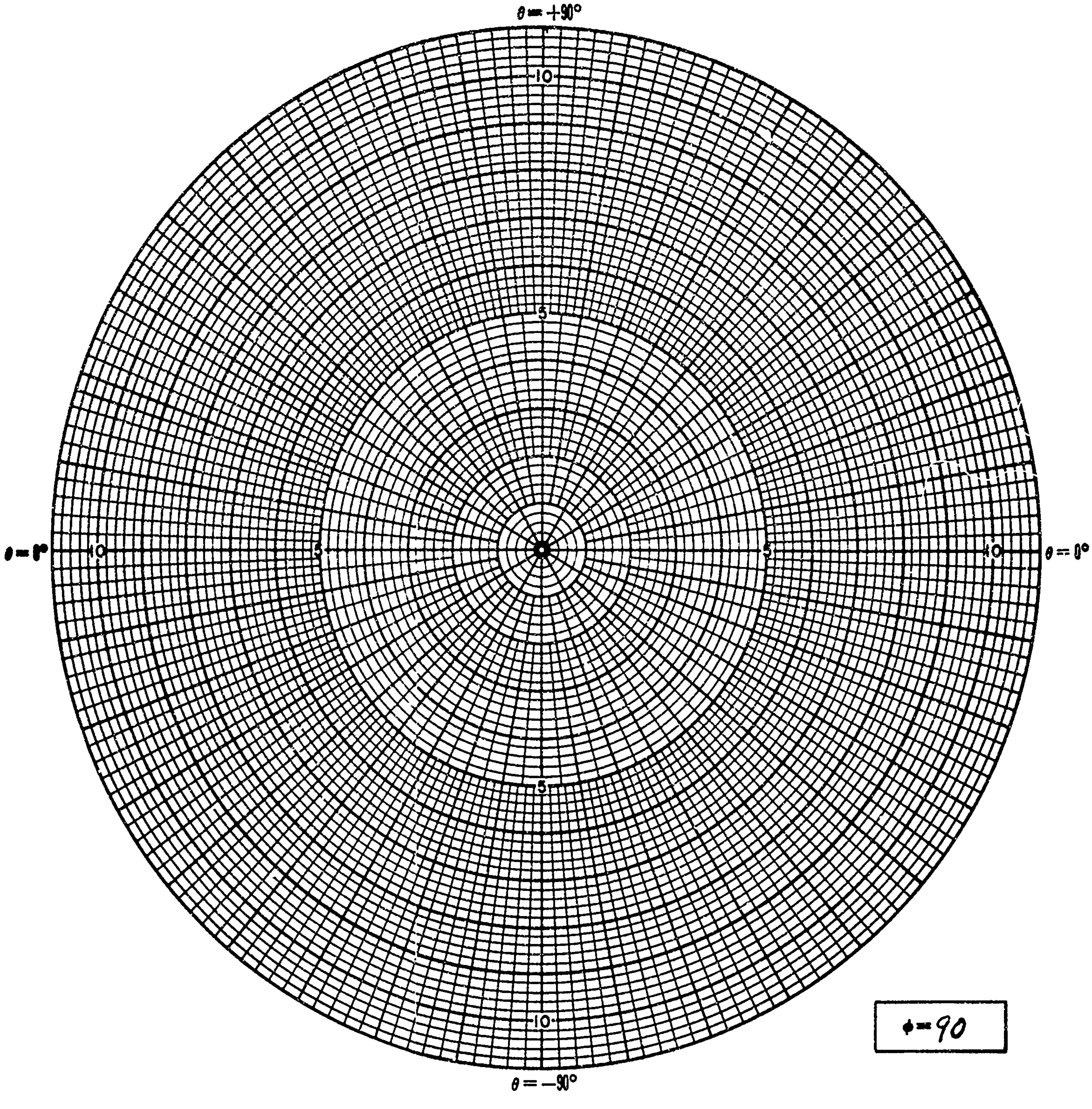


Figure 3-47. 136.11 MC Antenna Radiation Patterns (Sheet 3 of 3)

B

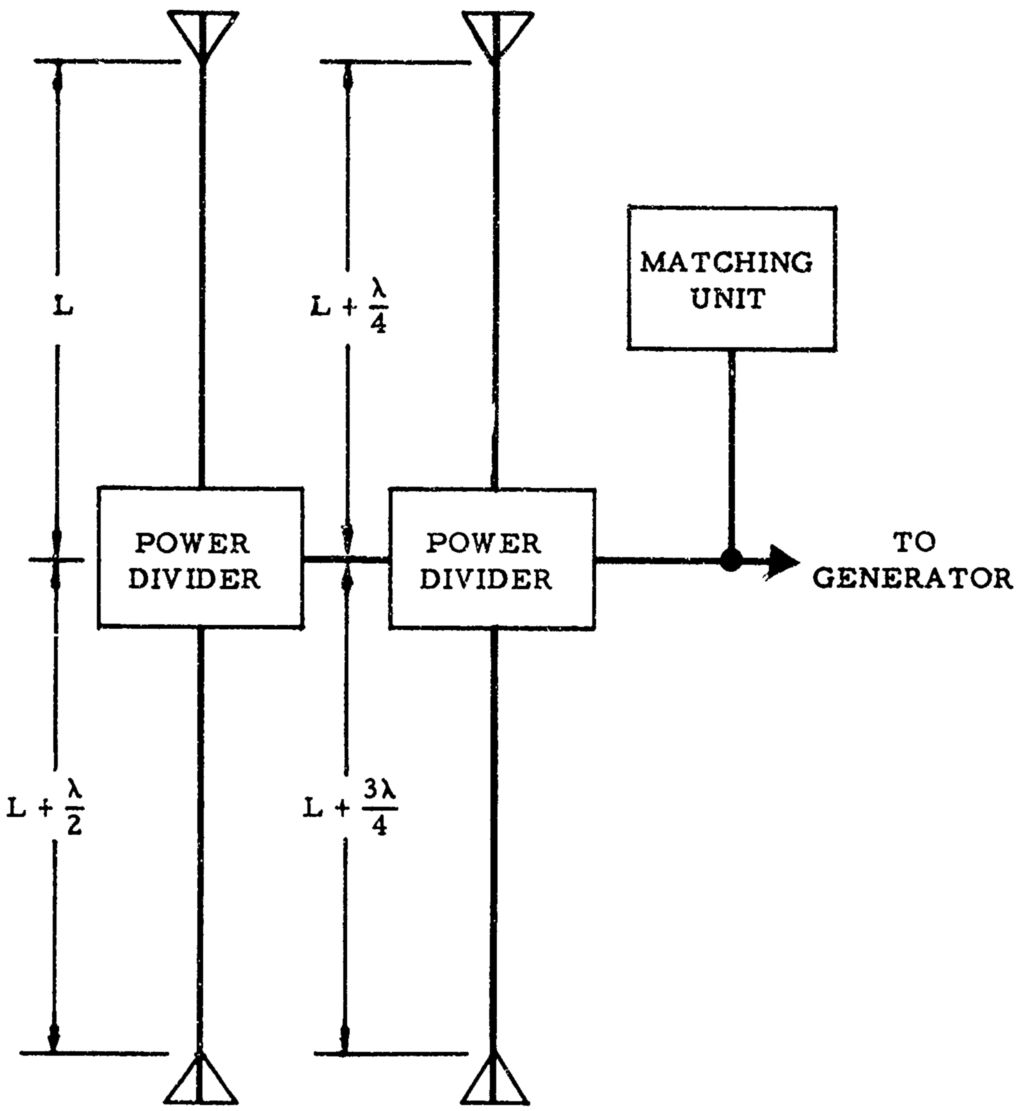


Figure 3-48. 224.5 MC Antenna Array Diagram

except for length, to the antenna systems used on the Navel Research Laboratory's series of satellites. This antenna system was chosen because of its simplicity and proven design. Several other types of antennas, such as the ones considered for the 420-450 mc array were investigated, but proved to be unfeasible for the spacecraft application.

3.8.4.2.1 The four 1/4-wavelength stubs are connected to the transponder through a cabling network which connects the stubs so that they form a turnstile antenna. The length of the cable to each successive antenna from the tee is increased by 1/4-wavelength. The initial length of cable is determined by the installation criteria. In this case, 10 inches was required to connect the antenna to the tee. This resulted in the following lengths of cable:

(1) $L = 10$ inches

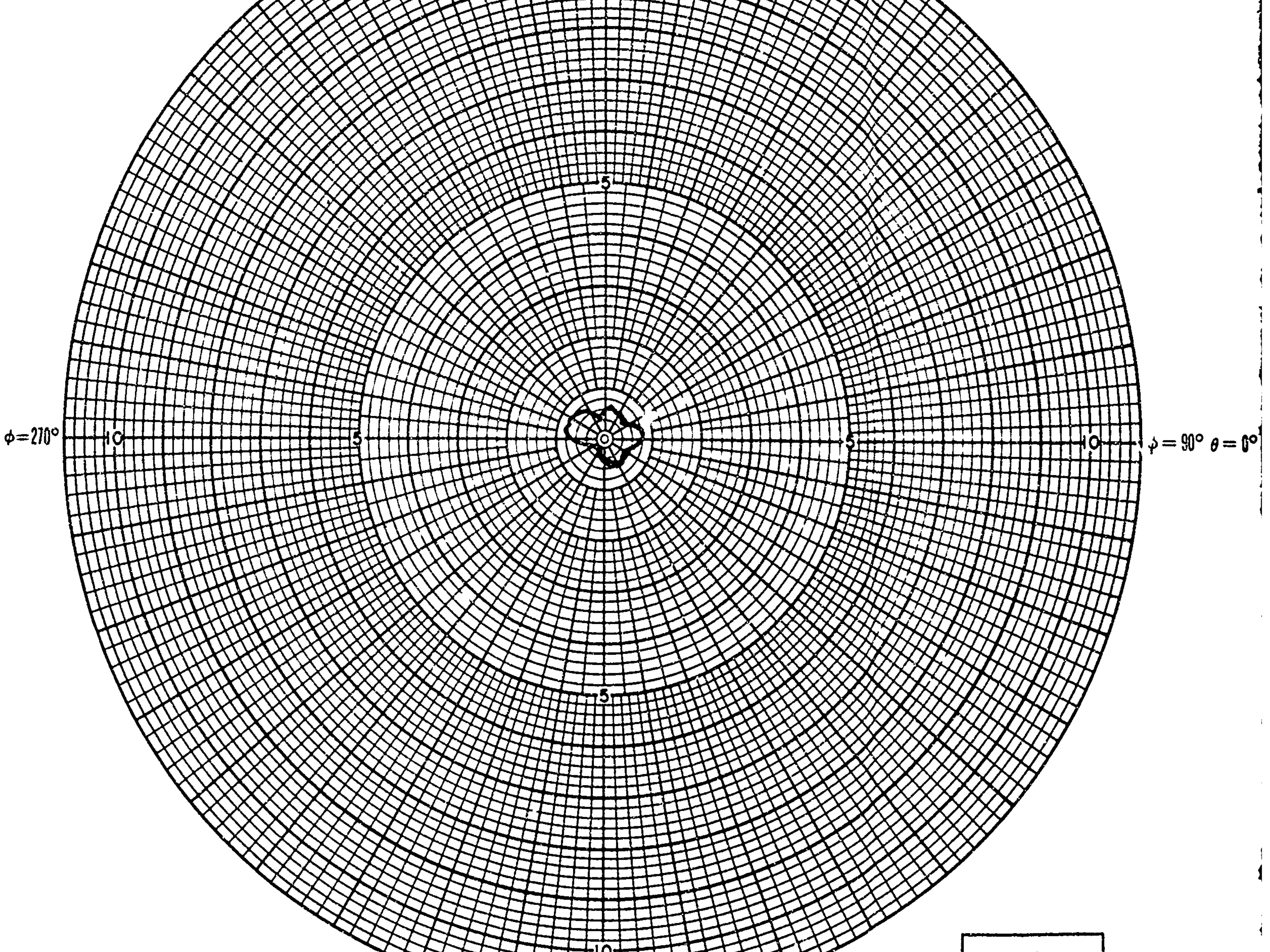
(2) $L + 1/4\lambda = 19.11$ inches

(3) $L + 1/2\lambda = 28.275$ inches

(4) $L + 3/4\lambda = 37.385$ inches

A matching network was used to balance the impedance of the antenna cables. The impedances before and after the insertion of the matching unit for Spacecraft No. 1 are shown on the Smith chart in figure 3-54. The network consisted of a 5.8-inch open-ended stub located 2.9 inches from the center of the tee toward the transmitter. The test equipment utilized to measure the impedances is listed in table X. Figures 3-55 and 3-56 are plots of impedance vs frequency for Spacecrafts No. 2 and No. 3 after the matching networks were inserted. Table XI is a list of the data collected during the measurement of impedance vs frequency for Spacecraft No. 3.

3.8.4.2.2 This antenna system provides nearly omnidirectional coverage. Figure 3-57 is a pictorial view of the radiation pattern from the antennas. This pattern can best be described as two doughnuts perpendicular to each other. The basic element of the antenna, the dipole, normally radiates no energy in the cross-polarization plane. However, as the shell of the Spacecraft is large with respect to a wavelength, the ideal antenna pattern is



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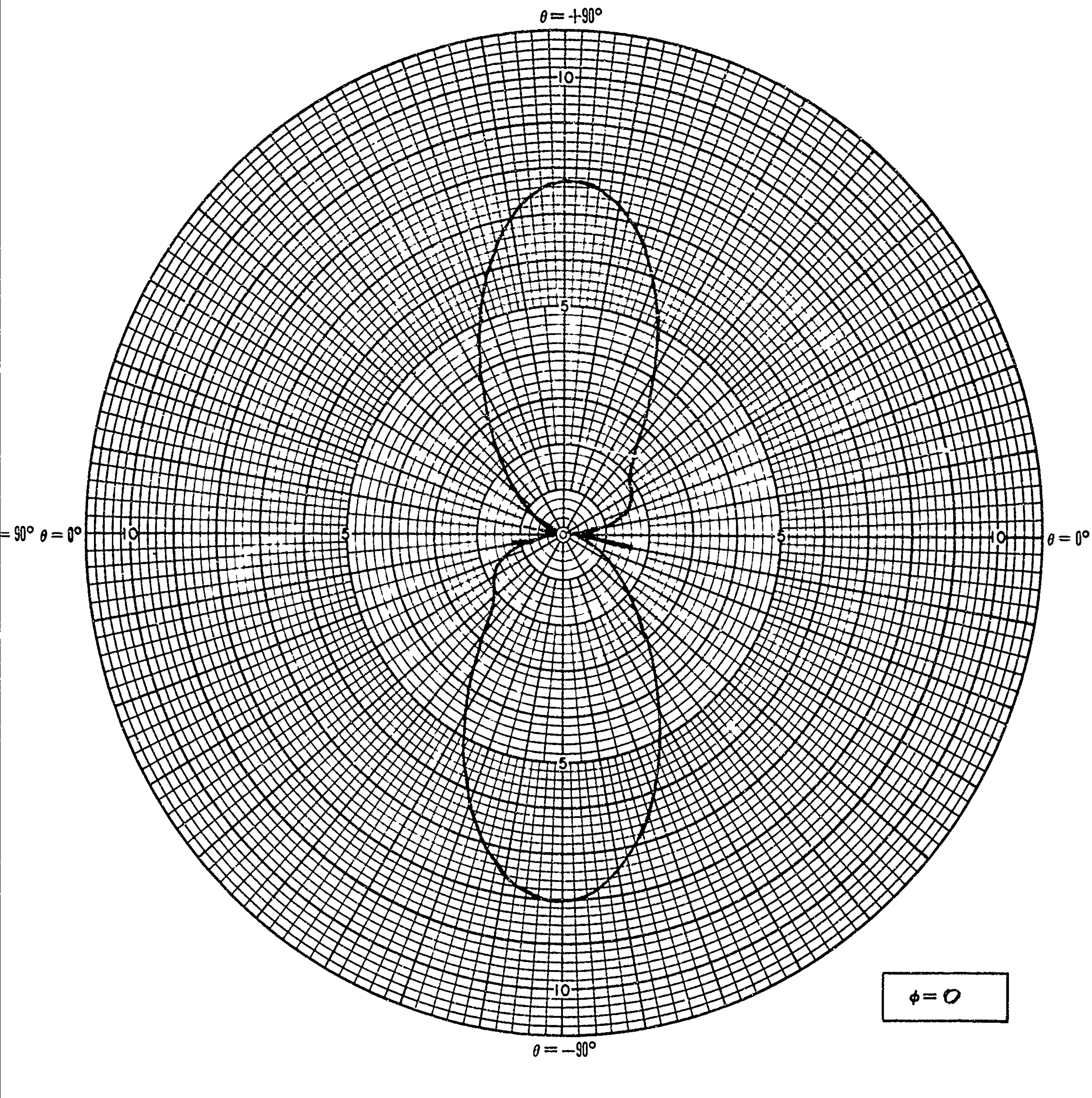
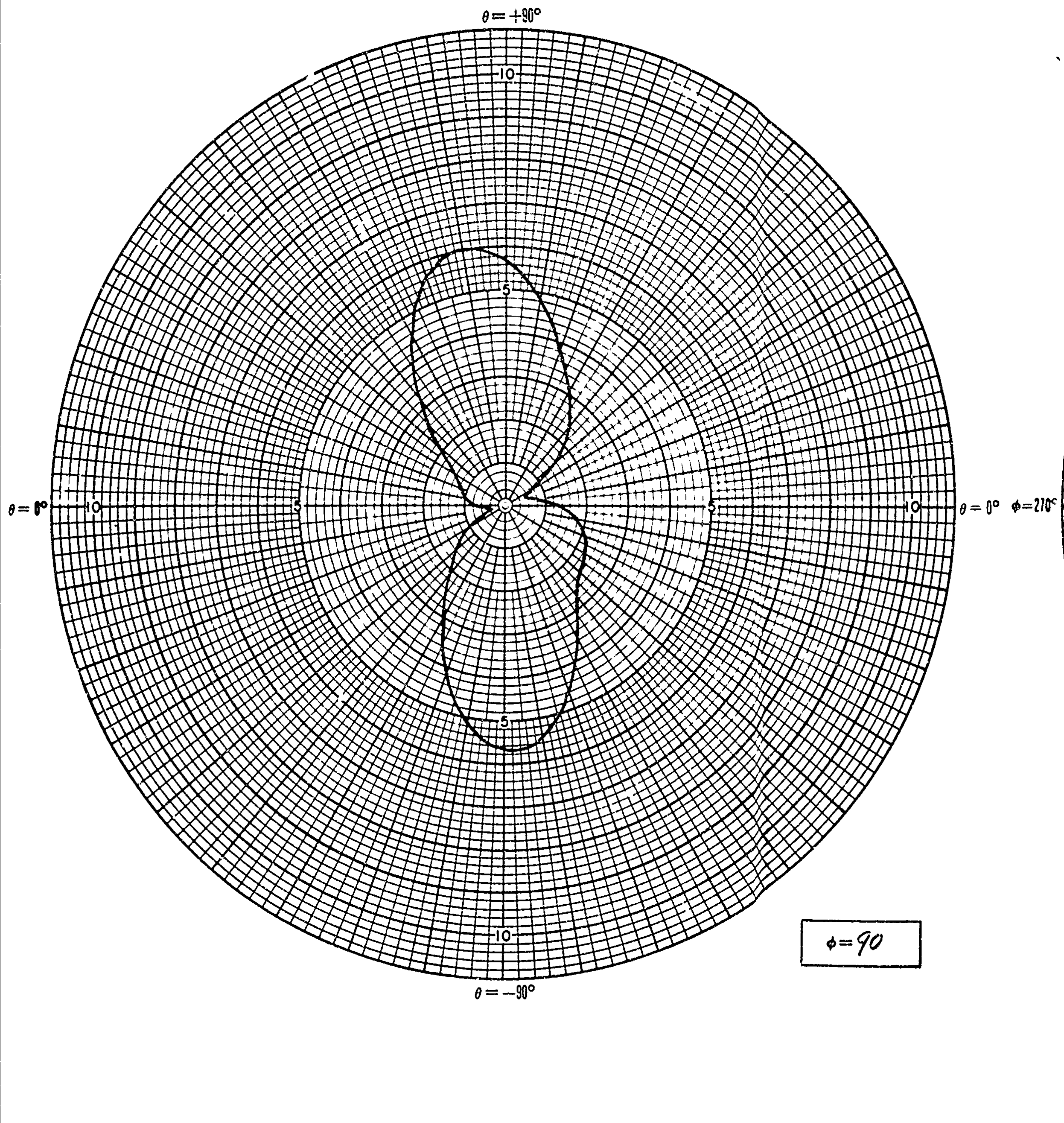


Figure 3-49. 224.5 MC Antenna Radiation Patterns (Sheet 1 of 3)

B



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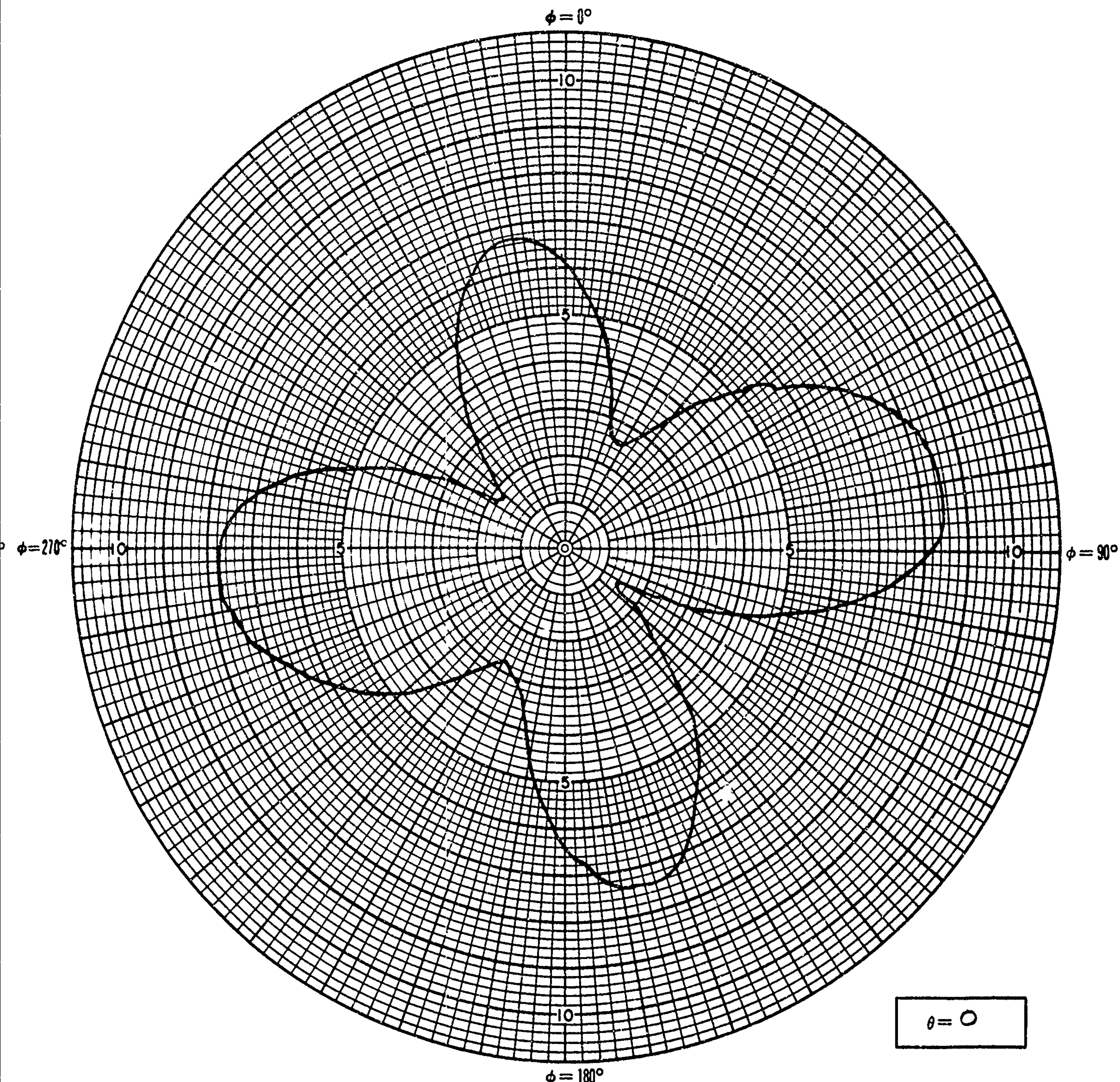
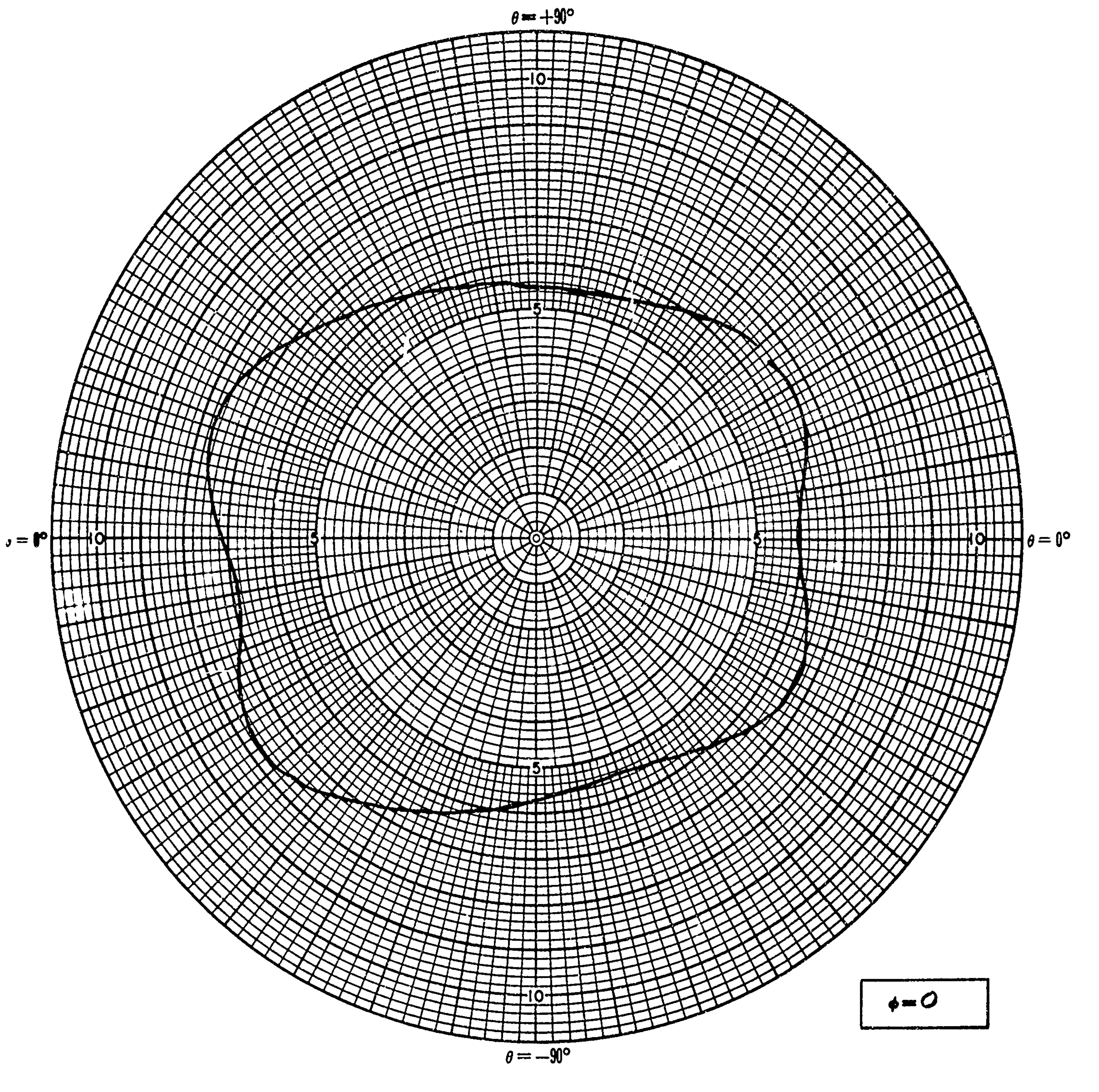


Figure 3-49. 224.5 MC Antenna Radiation Patterns (Sheet 2 of 3)



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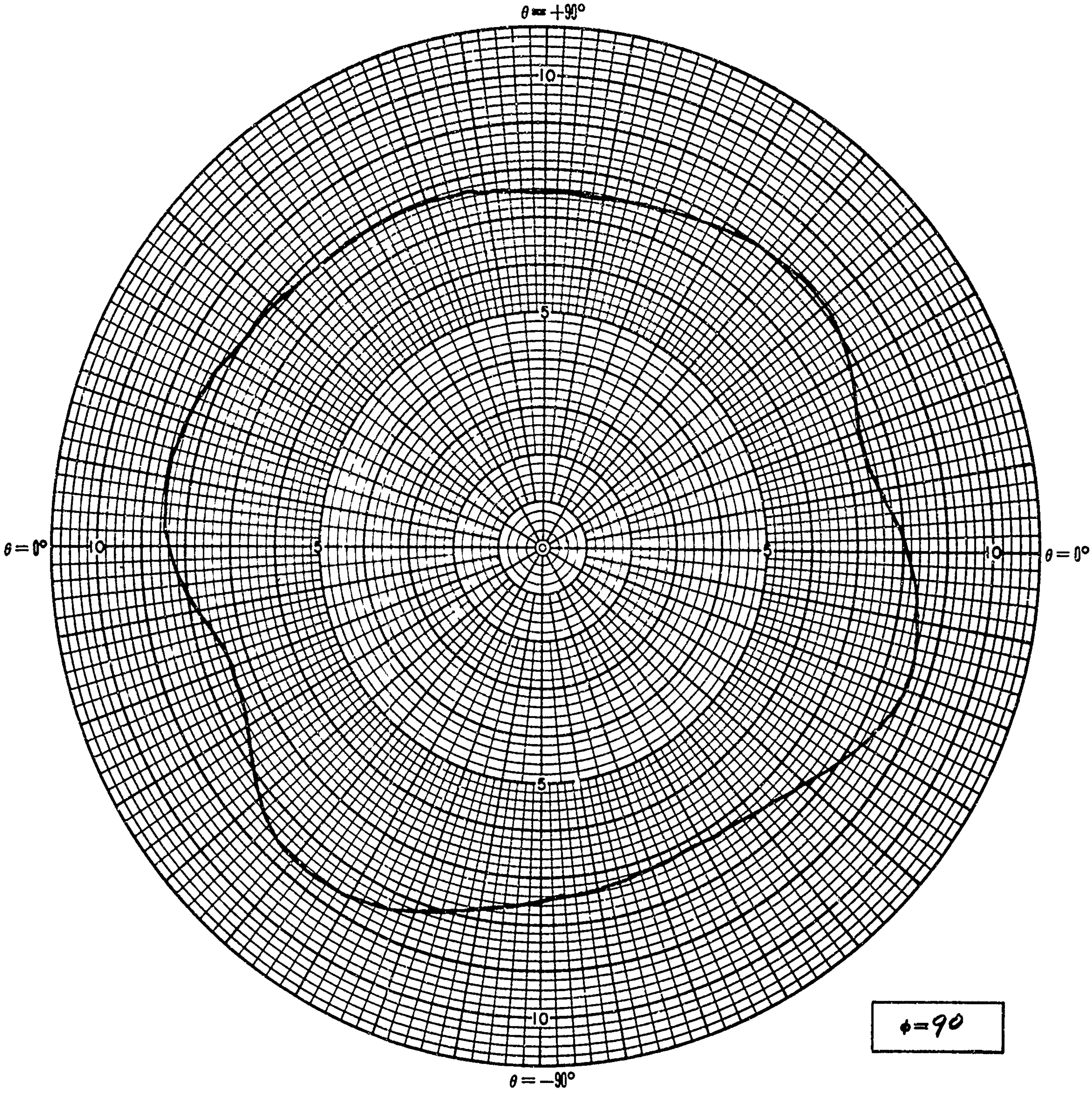


Figure 3-49. 224.5 MC Antenna
Radiation Patterns (Sheet 3 of 3)

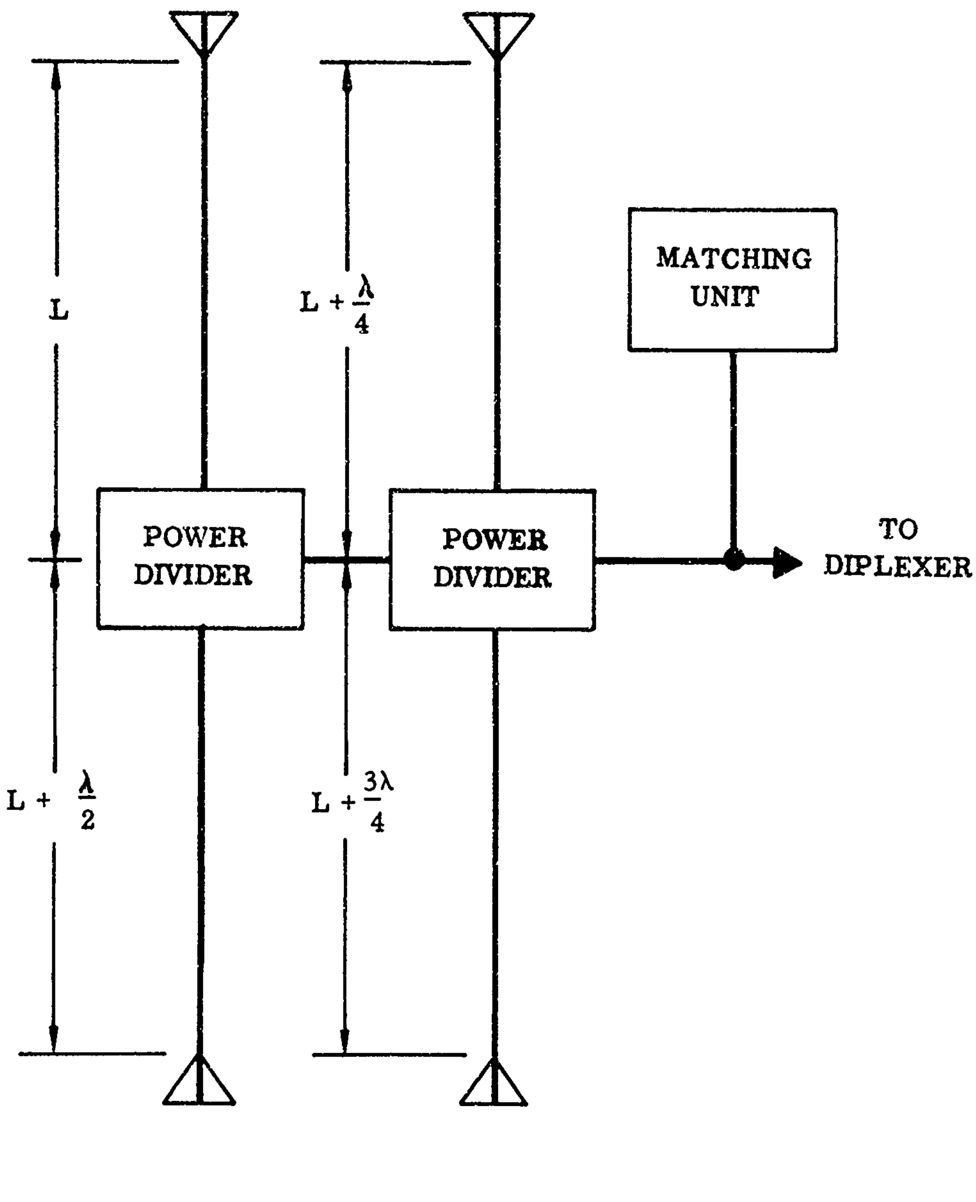
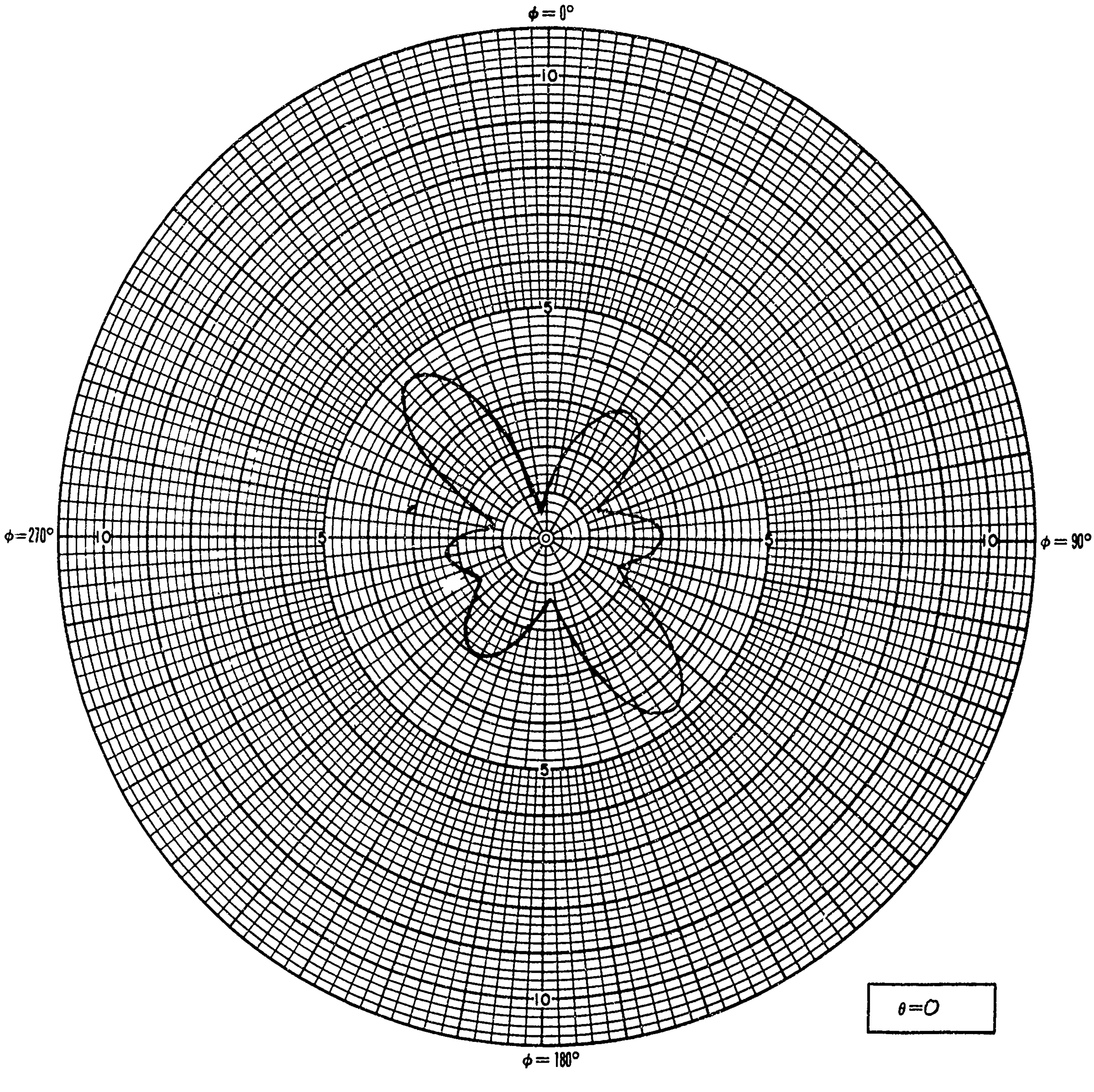


Figure 3-50. 421-449 MC Antenna Array Diagram



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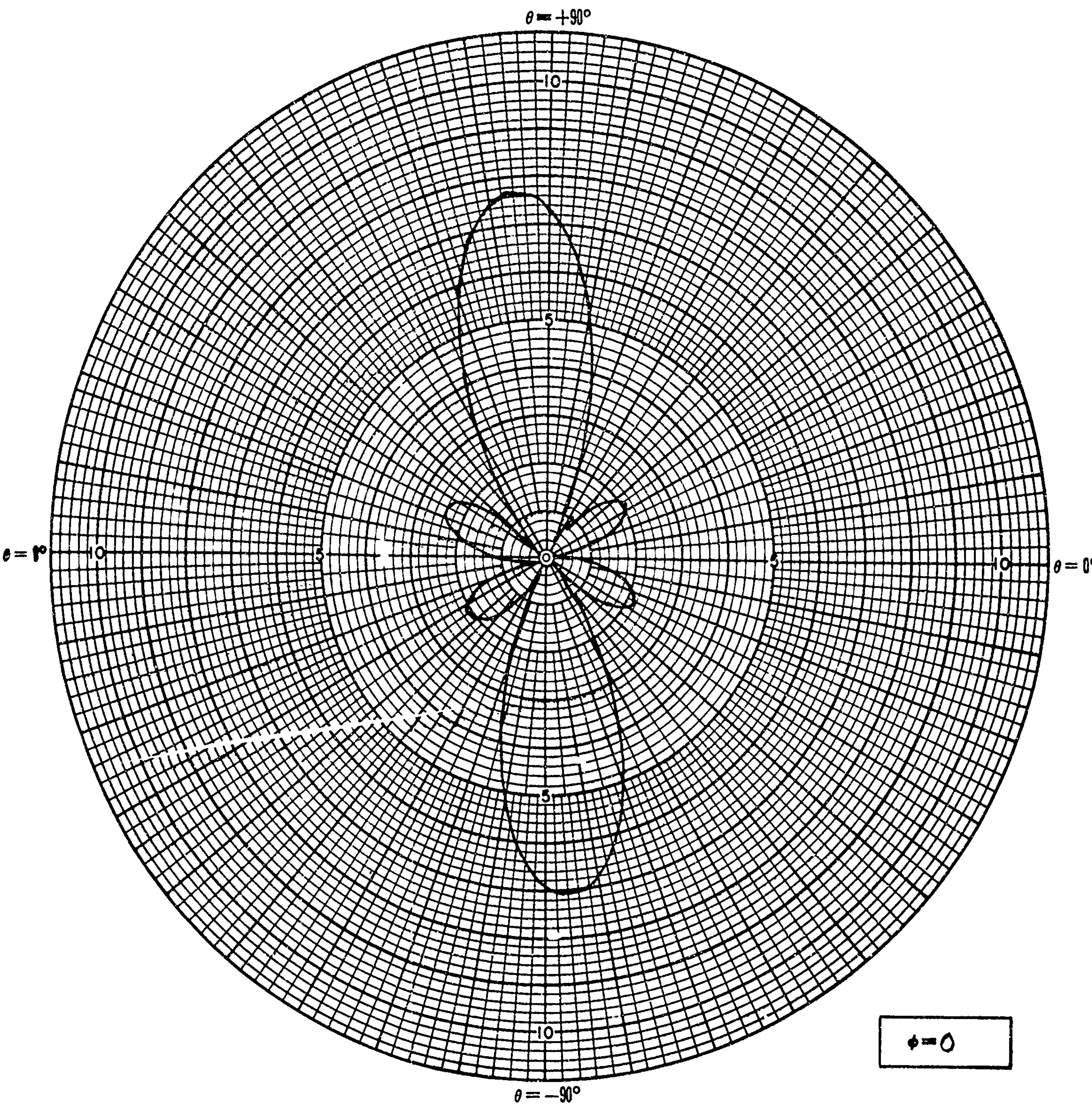
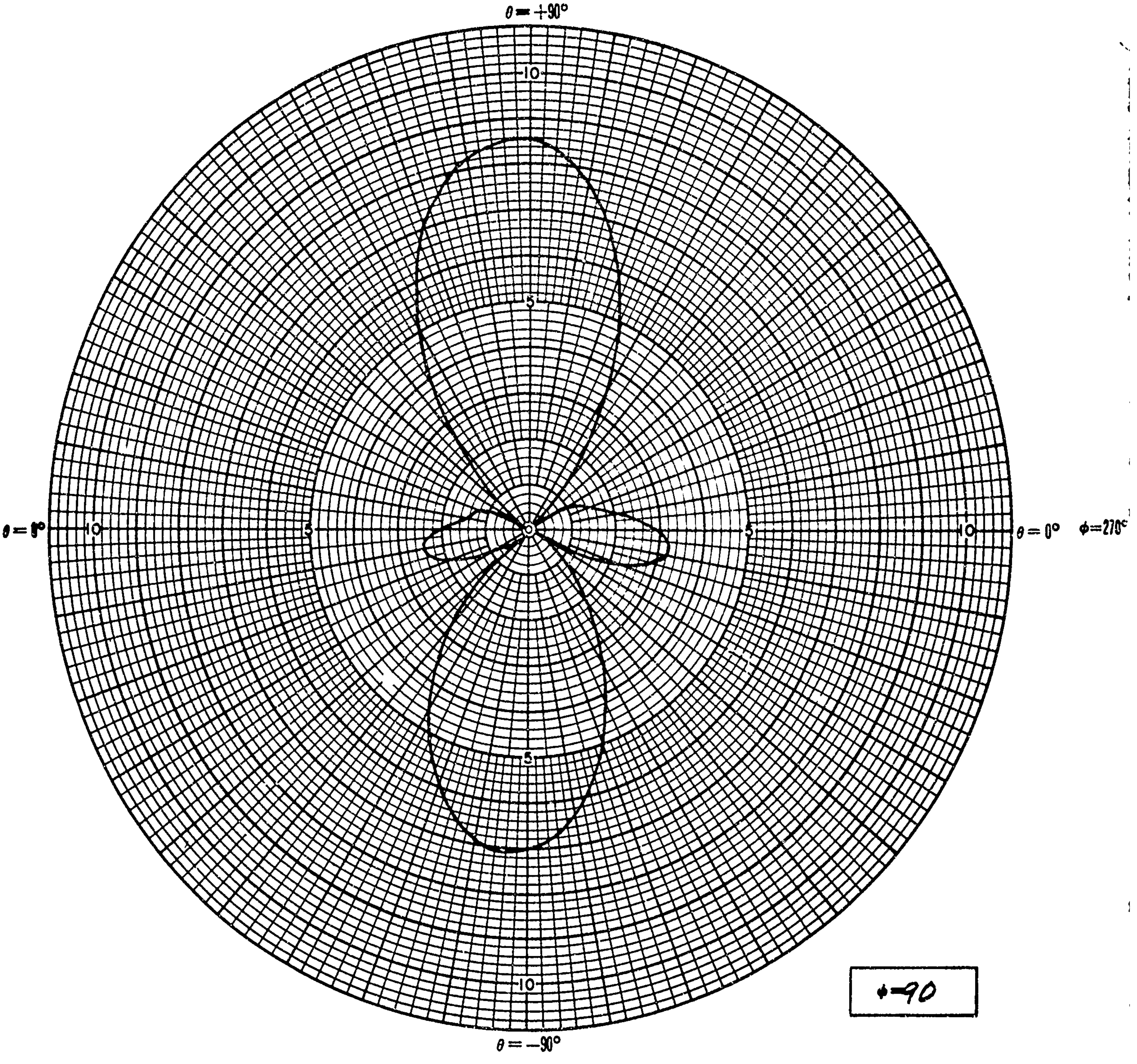


Figure 3-51. 421 MC Antenna Radiation Patterns (Sheet 1 of 3)



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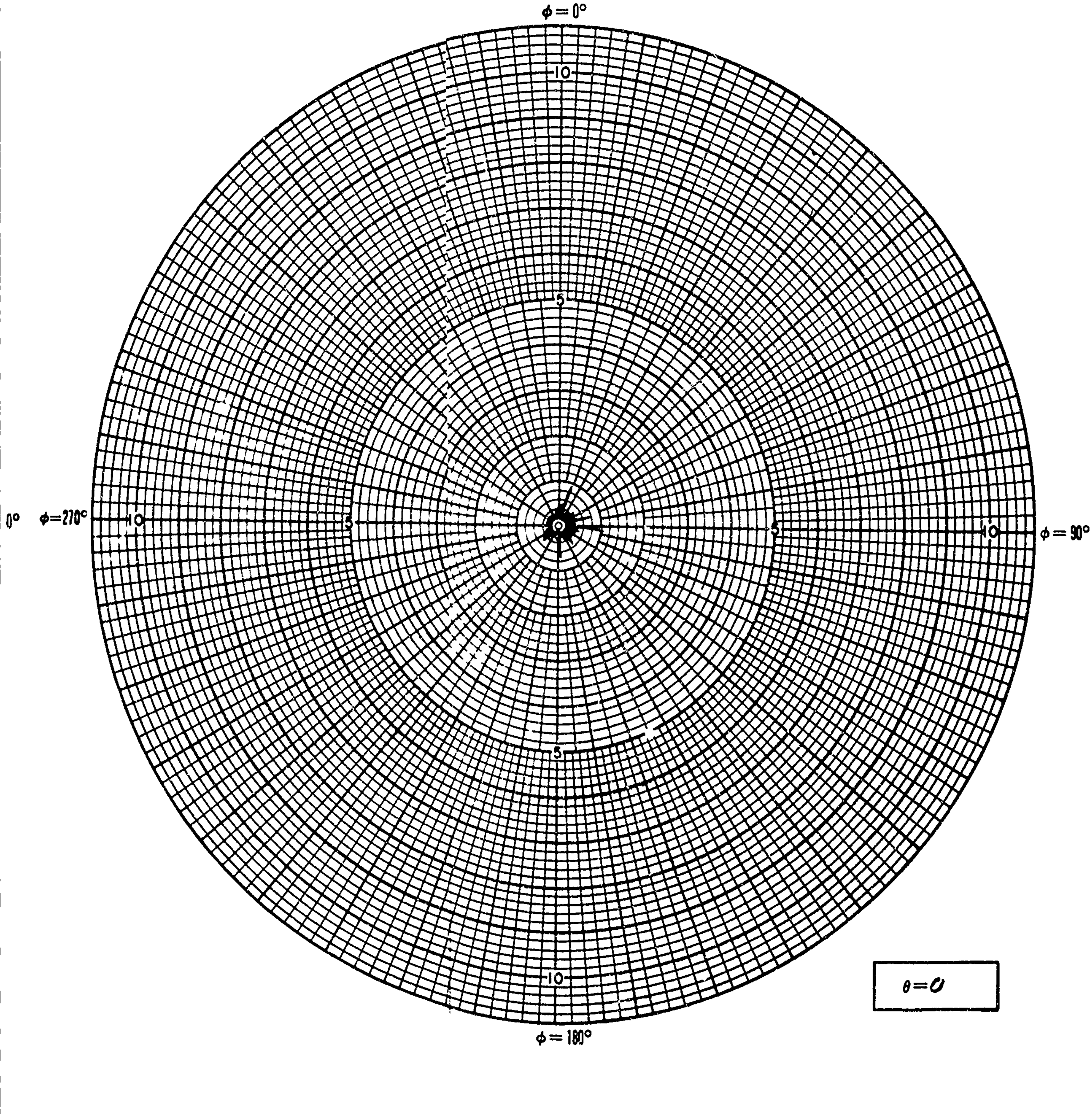
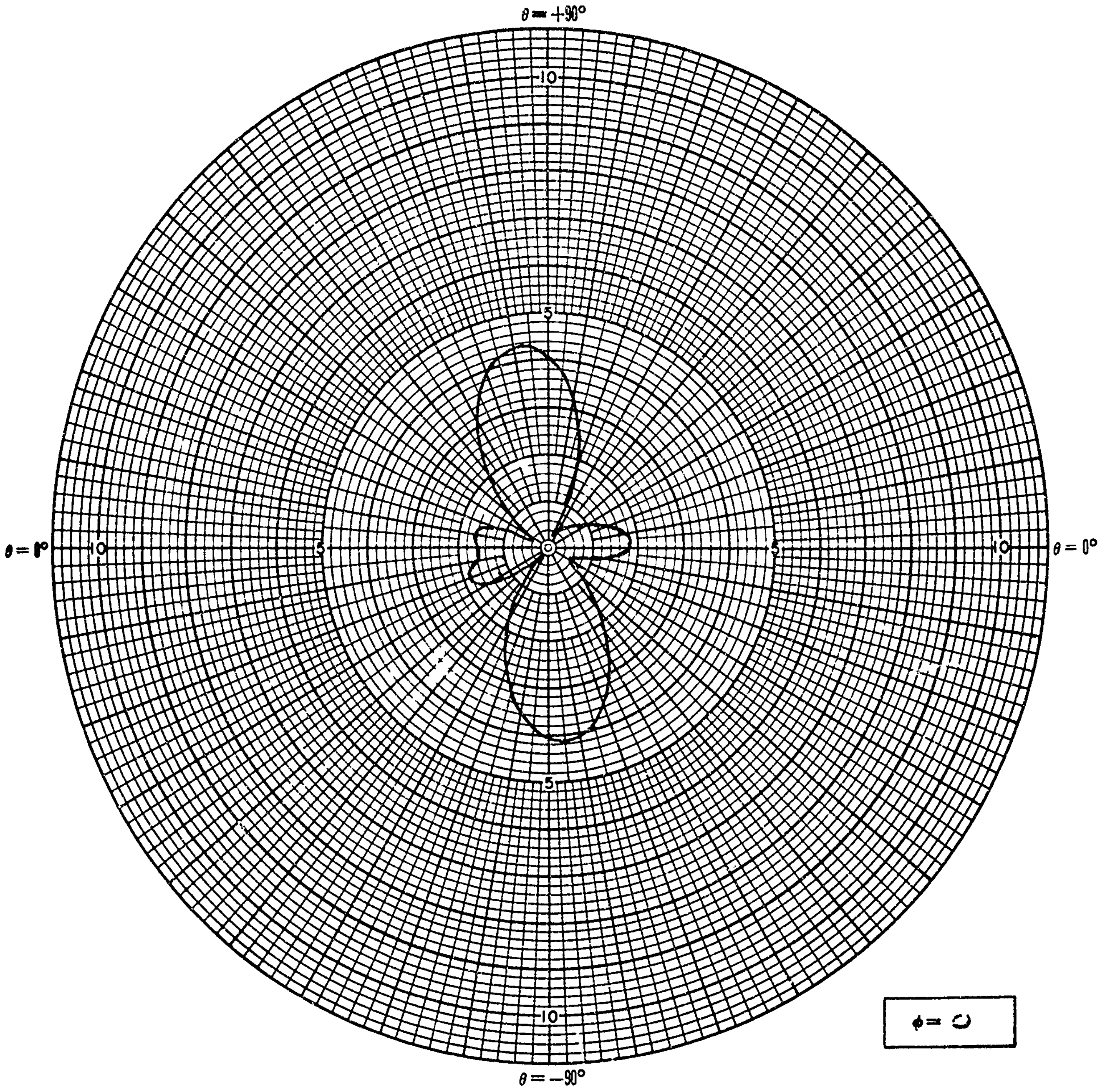


Figure 3-51. 421 MC Antenna Radiation Patterns (Sheet 2 of 3)

B



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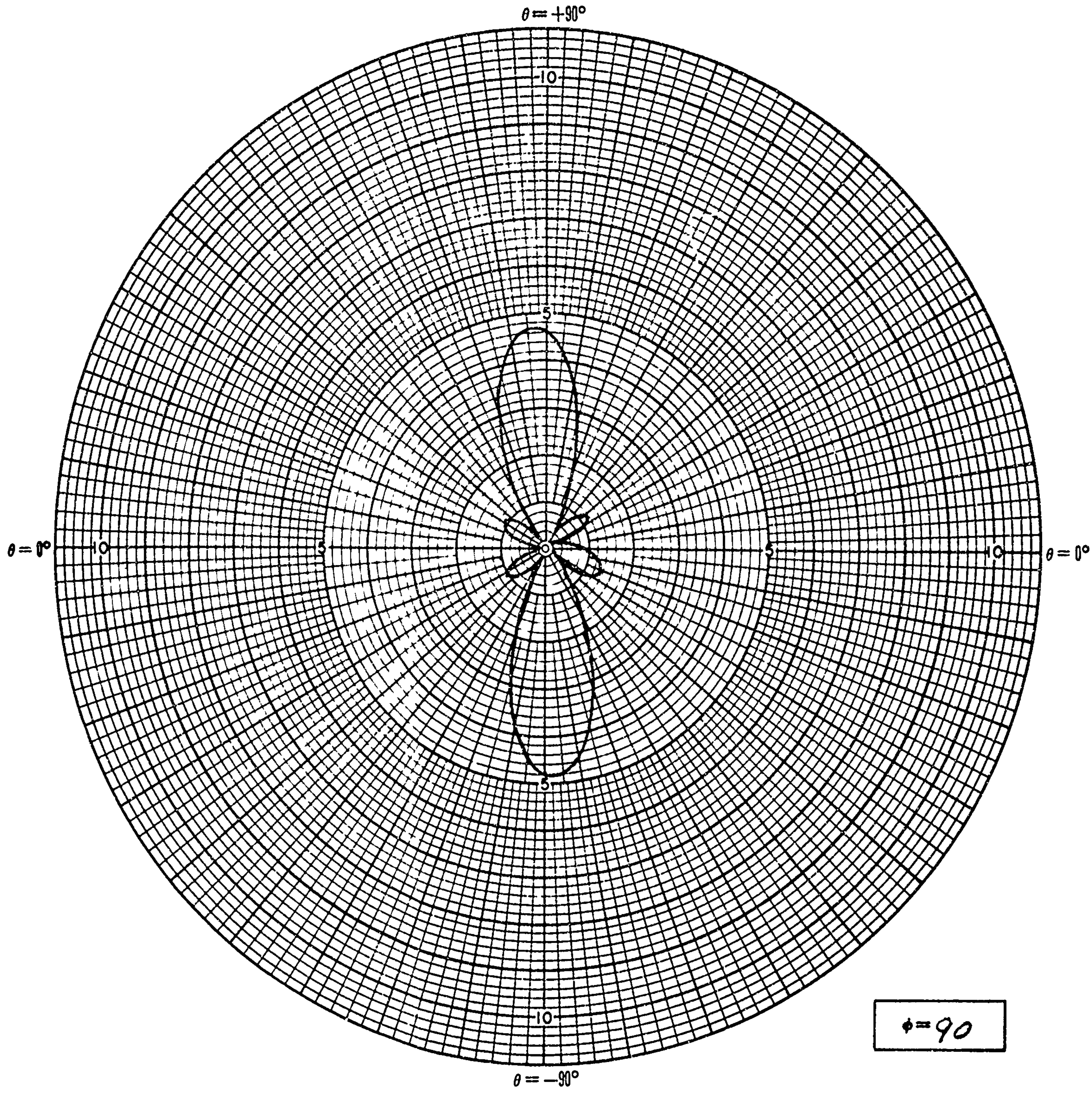
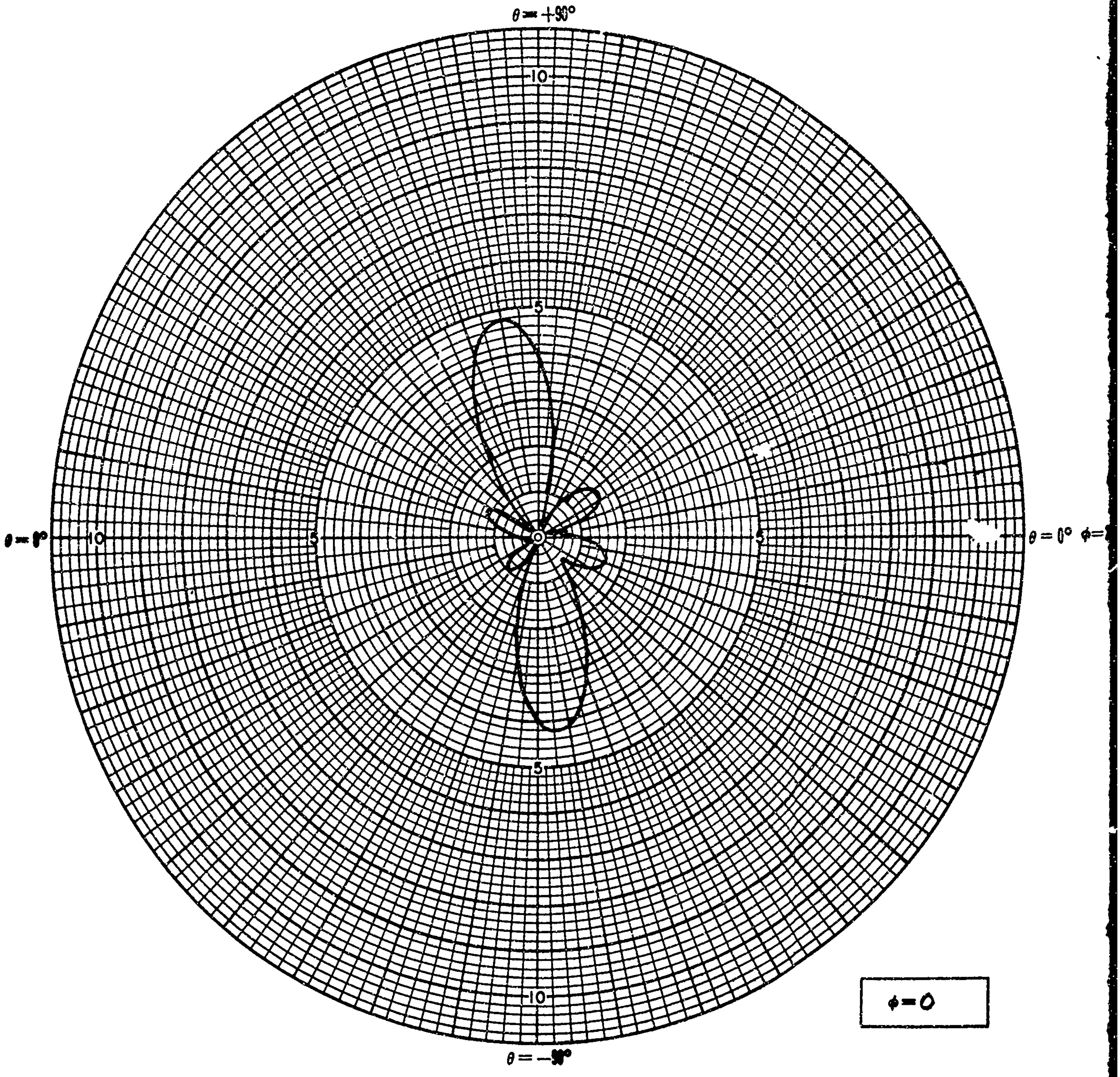


Figure 3-51. 421 MC Antenna Radiation Patterns (Sheet 3 of 3)

B



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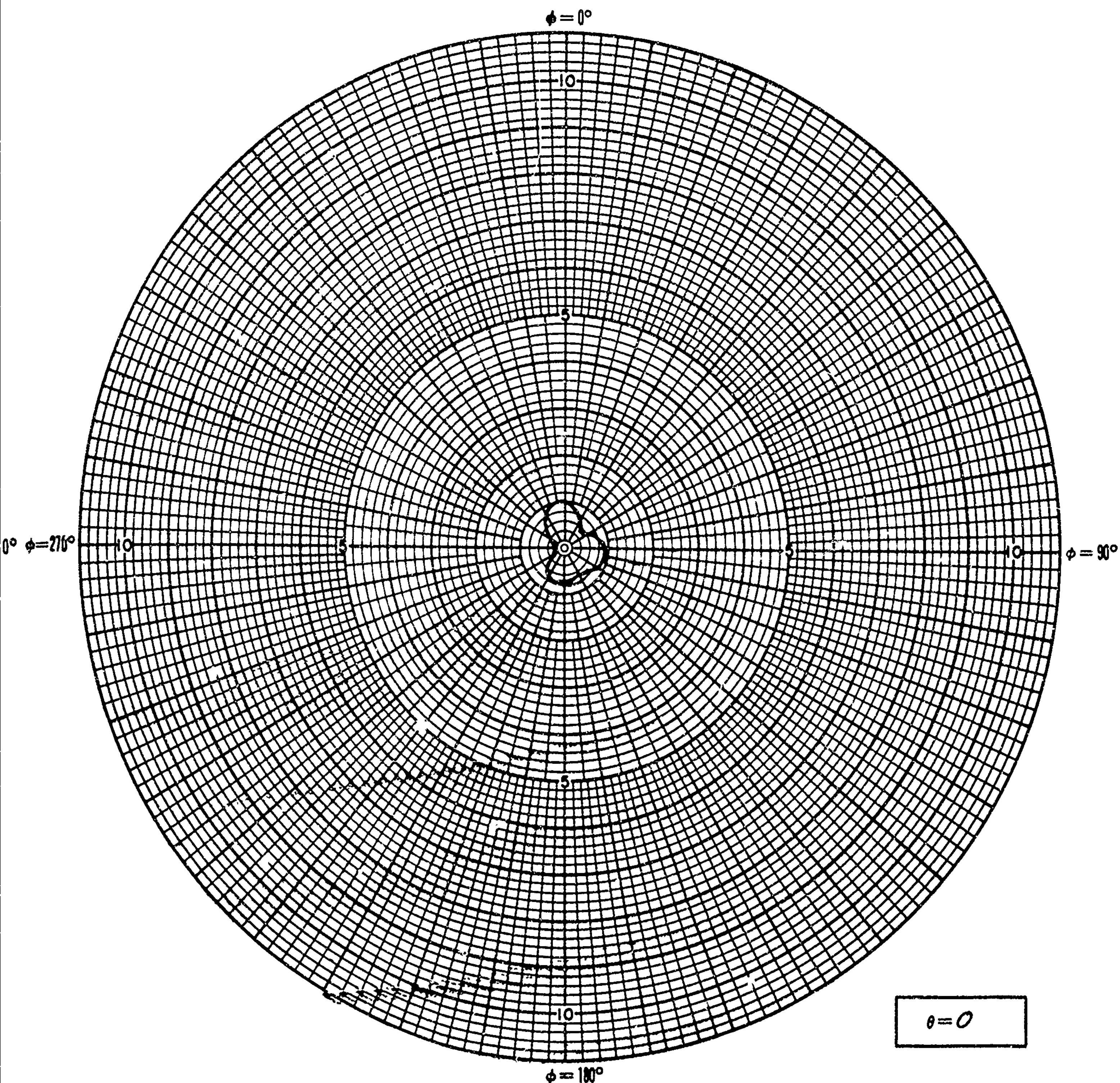
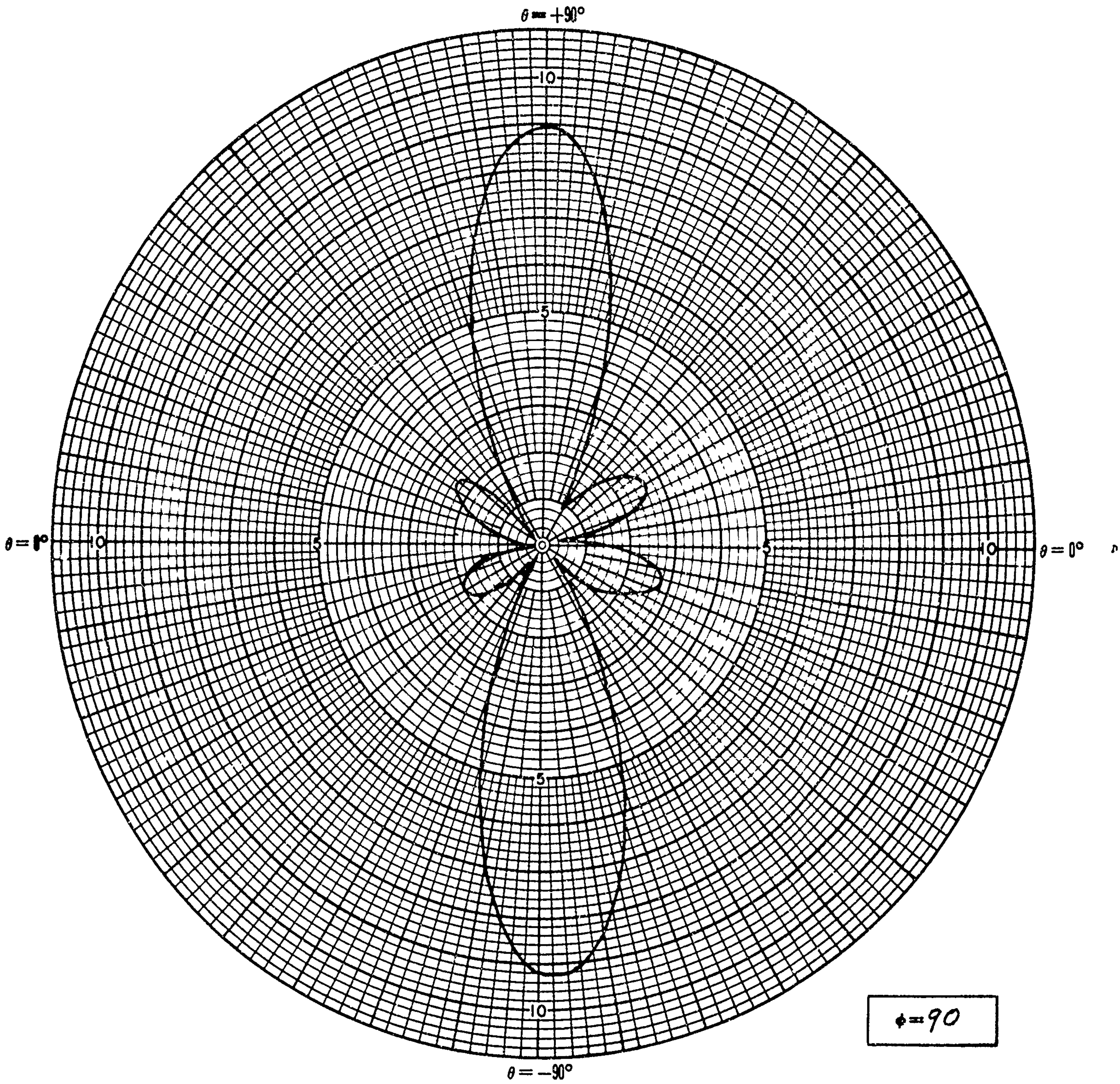


Figure 3-52. 449 MC Antenna Radiation Patterns (Sheet 1 of 3)



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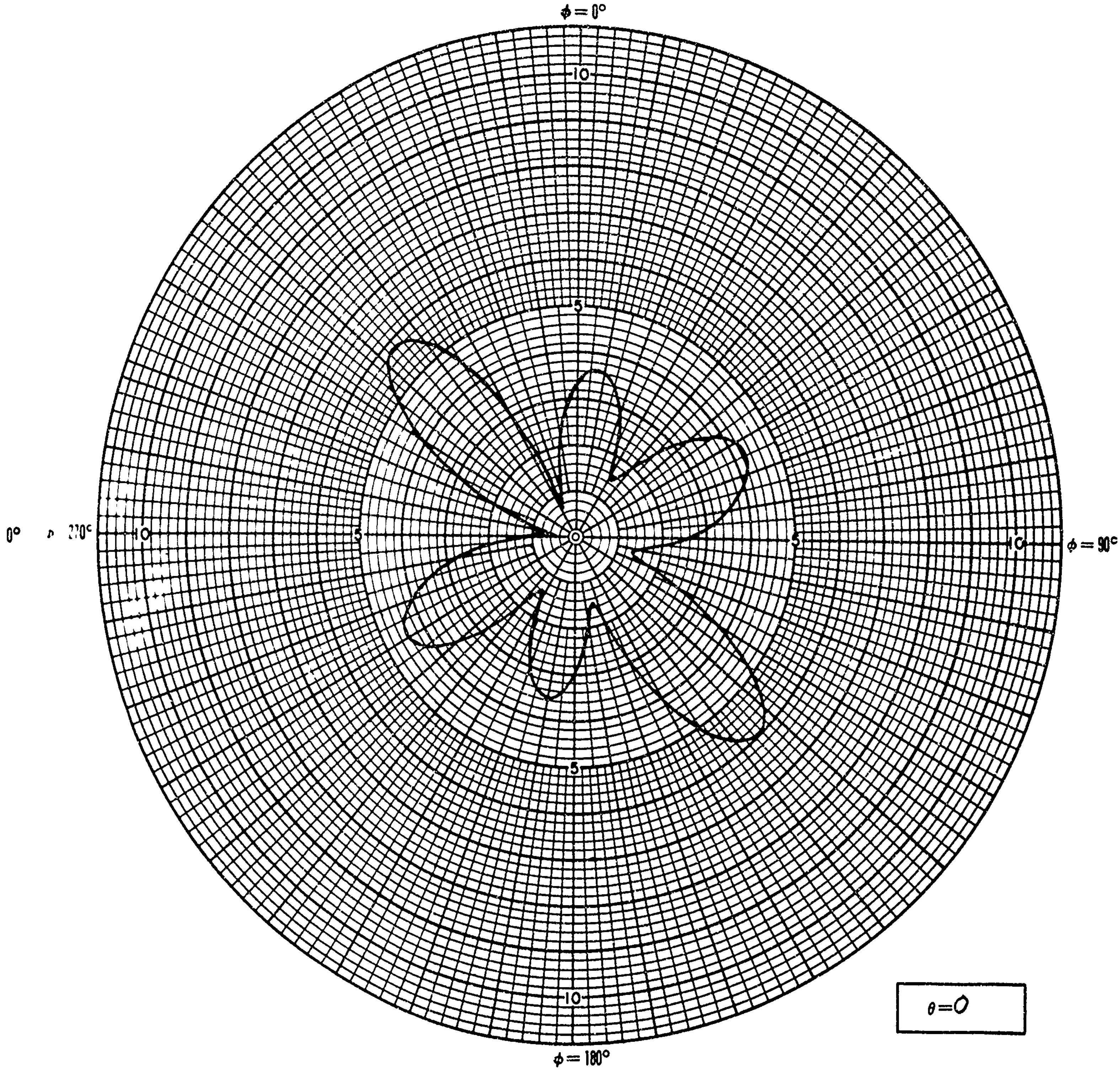
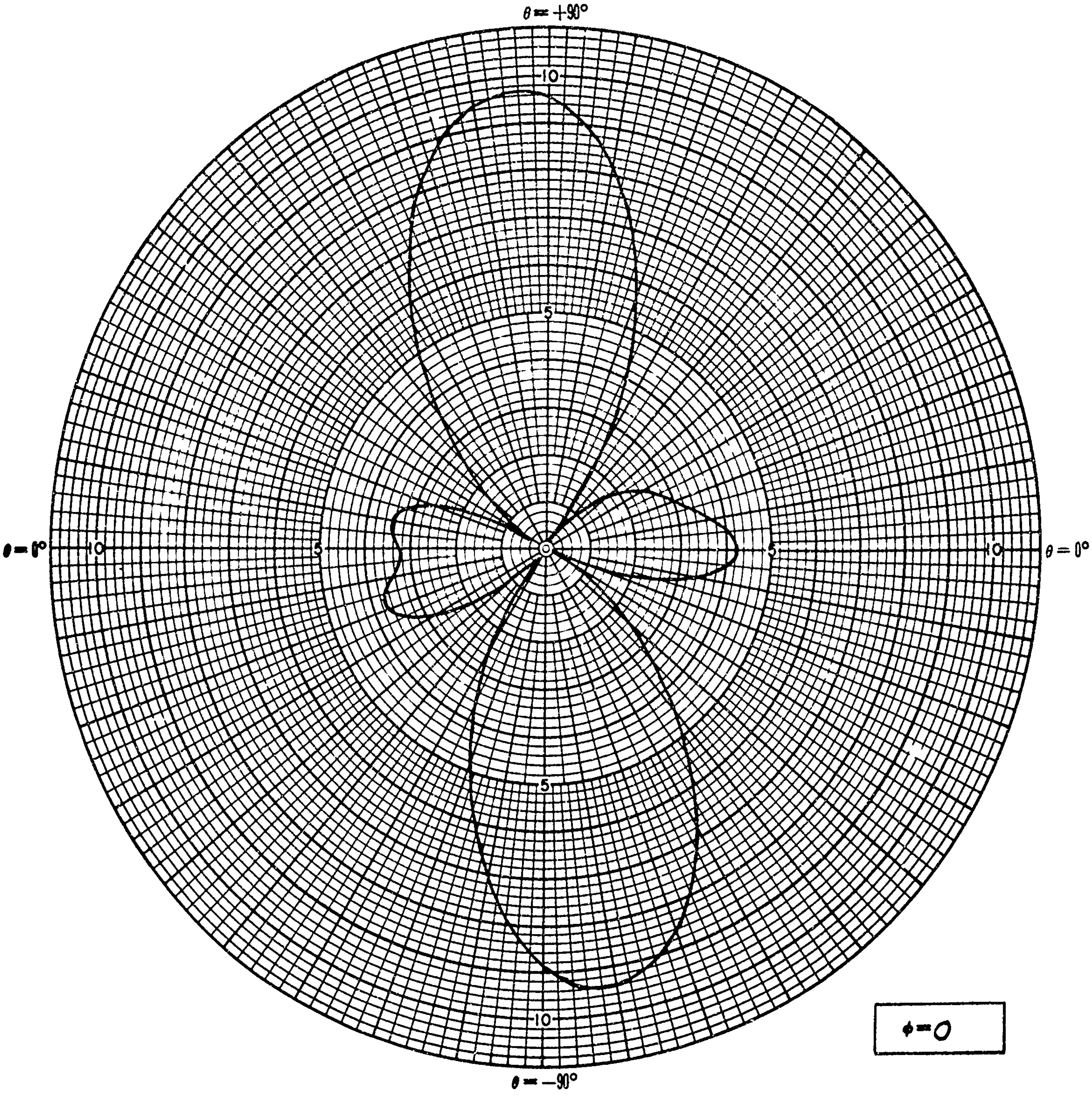


Figure 3-52. 449 MC Antenna Radiation Patterns (Sheet 2 of 3)



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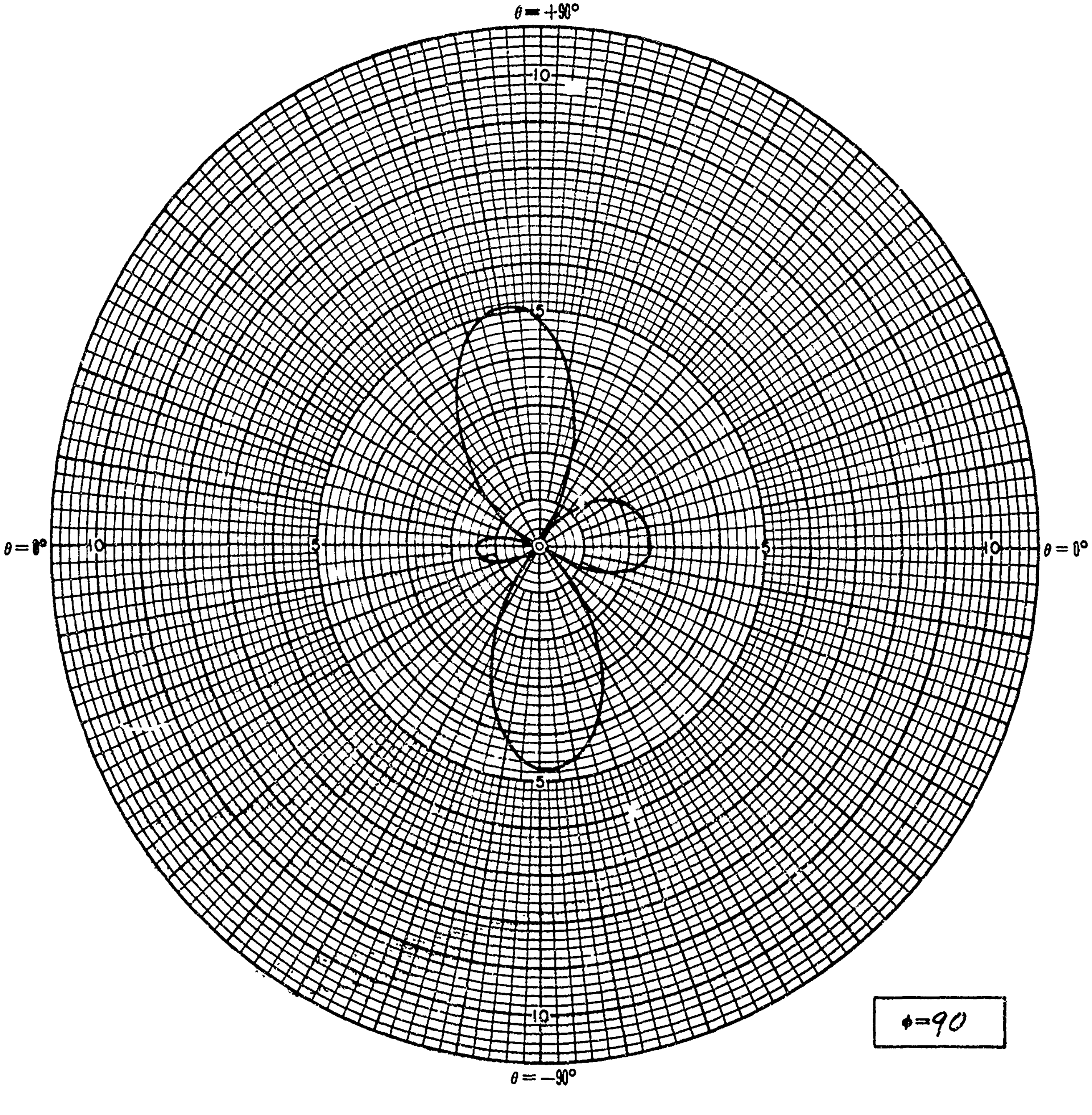


Figure 3-52. 449 MC Antenna
Radiation Patterns (Sheet 3 of 3)

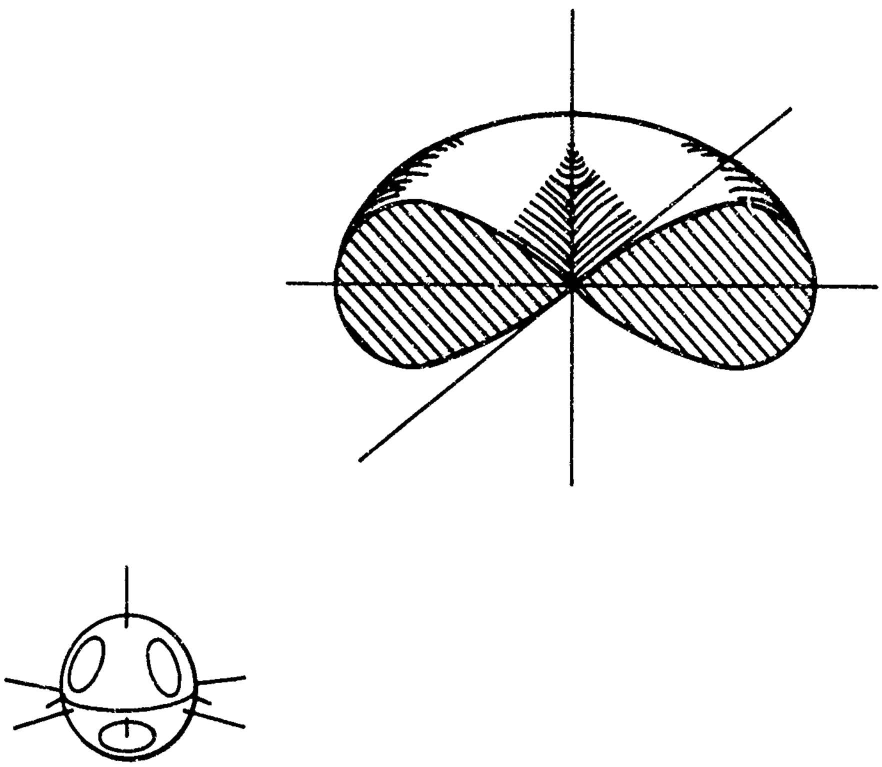


Figure 3-53. Cross Section of 136.11 MC Antenna Radiation Pattern

contaminated, and the shell provides some radiation. This energy fills in the nulls in the antenna patterns so that the deepest minima for any polarization at any aspect angle is only -16 db.

3.8.4.3 Development of 421-449 MC Antenna Array. The 421-449 mc antenna array is used in conjunction with the 1 watt 449 mc transmitter and the 421 mc receiver of the SECOR transponder. Four 1/4-wavelength dipoles are connected together to form a turnstile antenna. This antenna system, similar (except for length) to the 224.5 mc antenna systems, was chosen because of its simplicity and proven design. Several other types of antenna, such as the painted spiral, the cavity, and the spherical antenna were considered but proved to be unfeasible for the Geodetic Spacecraft. The painted spiral antenna, as used by Transit could not be used because of interference with the solar cells. The cavity-type antennas could not be used because of their large physical size at these frequencies. The spherical antenna would be formed by insulating the two halves of the shell from each other. This was impractical because it would have required a redesign of the mechanical structure.

3.8.4.3.1 The four 1/4-wavelength stubs are connected to the transponder through a cabling network which connects the stubs so that they form a turnstile antenna. The length of the cable to each successive antenna from the tee is increased by 1/4-wavelength. The initial length of cable is determined by the installation criteria. In this case 16 inches was required to connect the antenna to the tee. This resulted in the following lengths of cable:

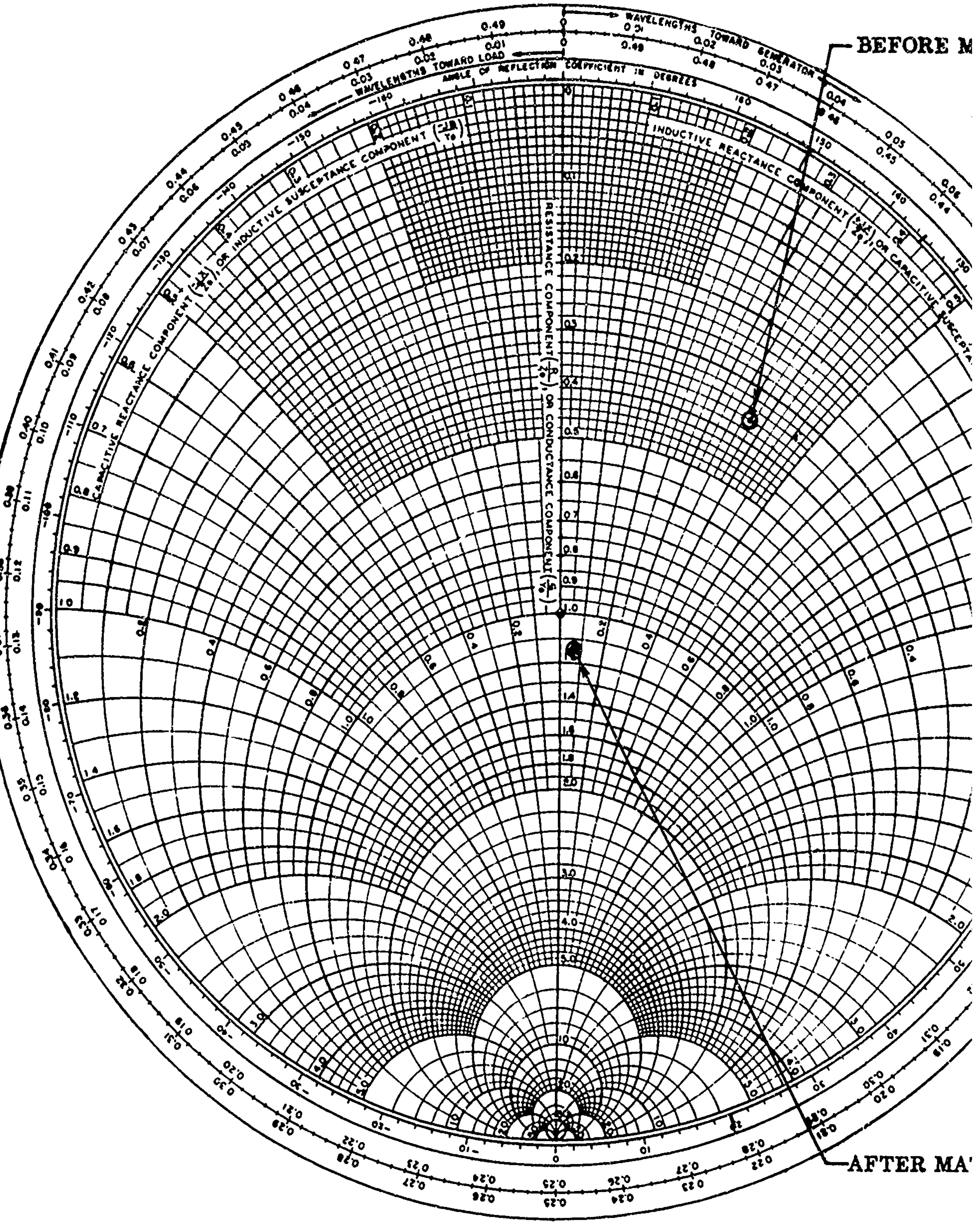
$$L = 16 \text{ inches}$$

$$L + 1/4 = 20.75 \text{ inches}$$

$$L + 1/2 = 25.50 \text{ inches}$$

$$L + 3/4 = 30.25 \text{ inches}$$

A matching network was used to balance the impedance of the antenna cables. The impedances before and after the insertion of the matching unit, for Spacecraft No. 1, are shown on the Smith chart in figure 3-58. The network consisted of a 2.9-inch open-ended stub connected directly to the tee. Refer to table X for a list of the test equipment


WAVELENGTHS TOWARD GENERATOR
WAVELENGTHS TOWARD LOAD
ANGLE OF REFLECTION COEFFICIENT IN DEGREES
INDUCTIVE REACTANCE COMPONENT
CAPACITIVE REACTANCE COMPONENT
OR INDUCTIVE SUSCEPTANCE COMPONENT
RESISTANCE COMPONENT (R/Zo), OR CONDUCTANCE COMPONENT (G/Yo)
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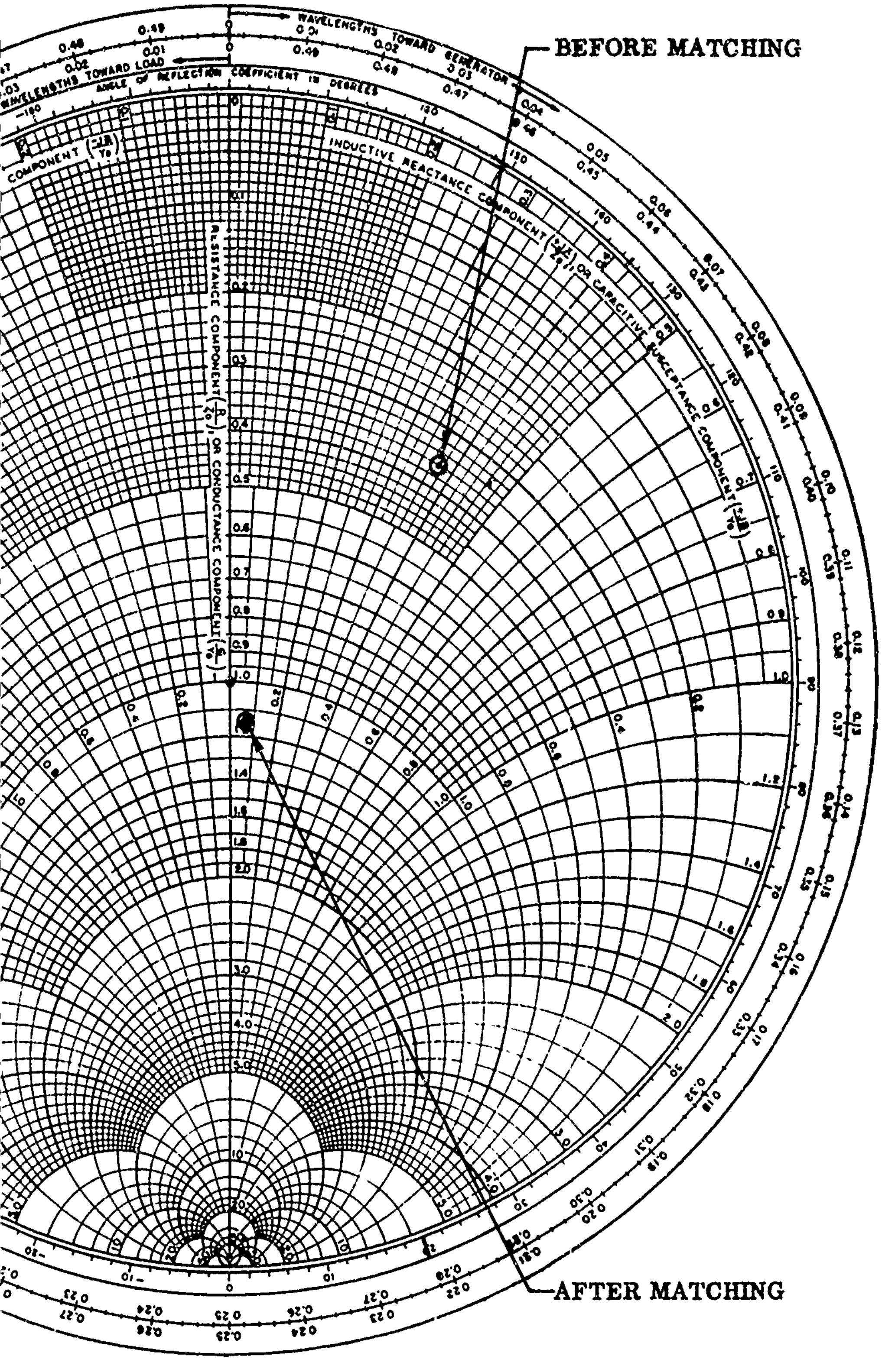
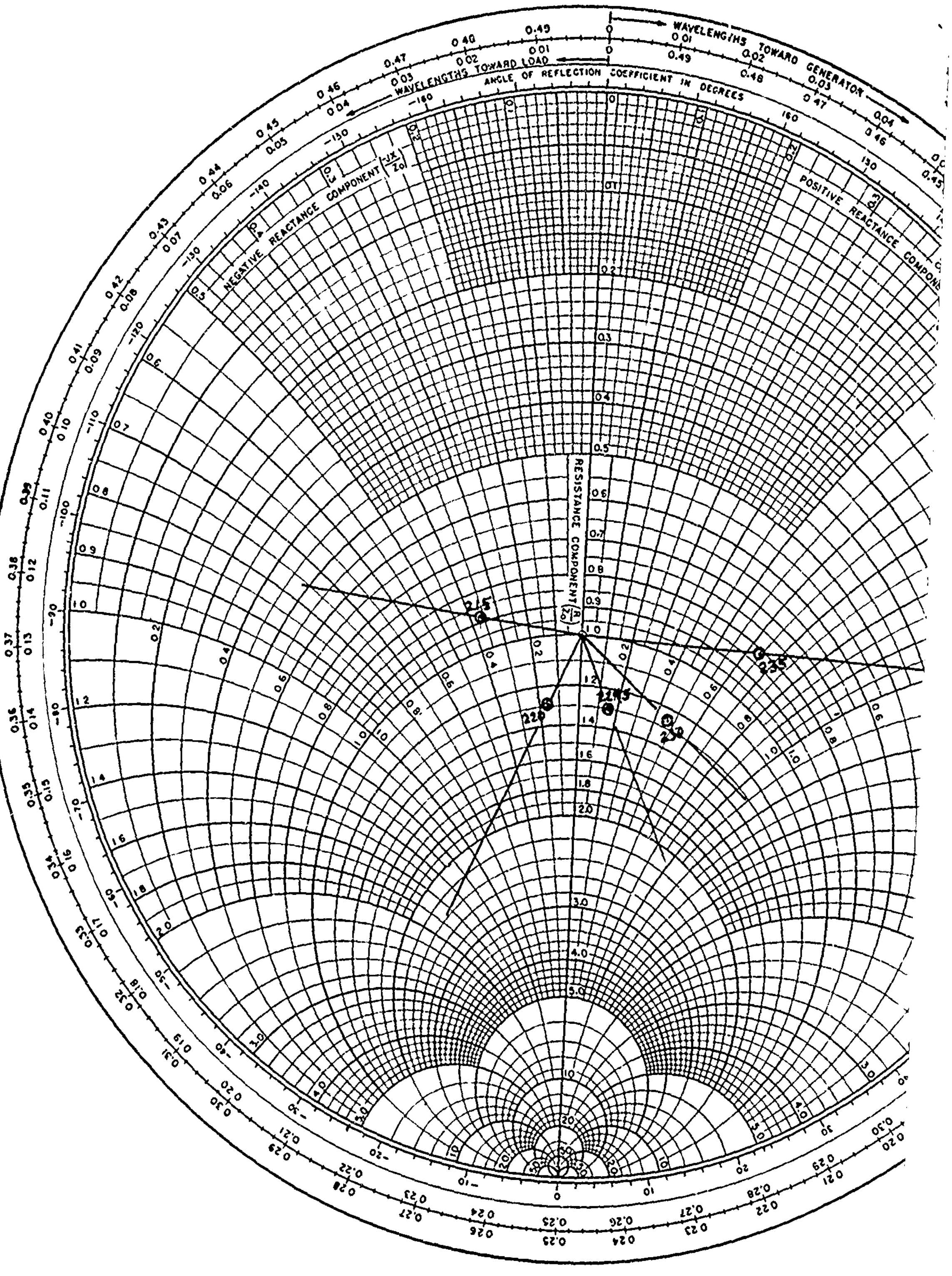


Figure 3-54. Smith Chart of 224.5 MC Antenna Impedance for Spacecraft No. 1

B



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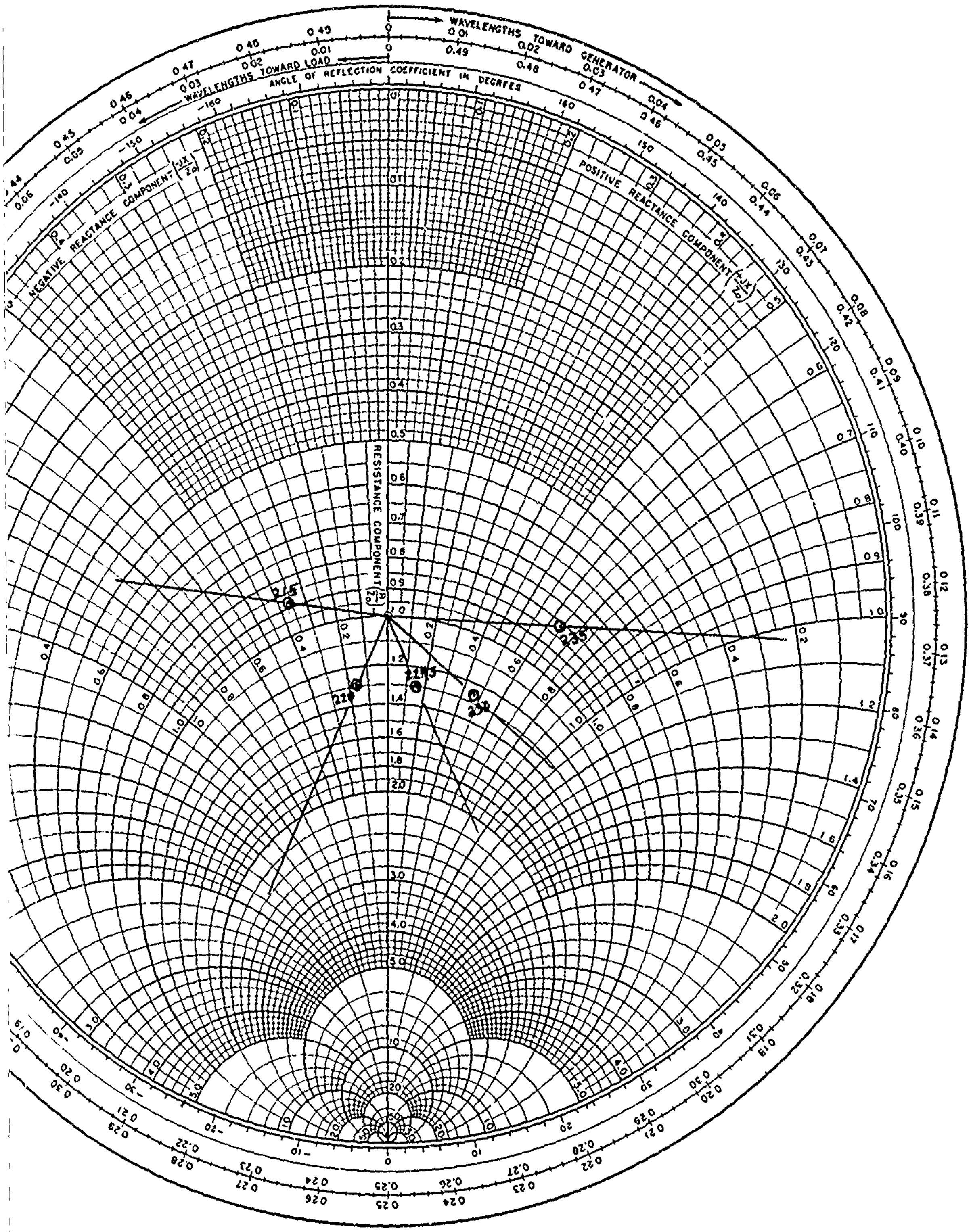


Figure 3-55. Smith Chart of 224.5 MC Antenna Impedance VS Frequency for Spacecraft No. 2 (After Matching)

B

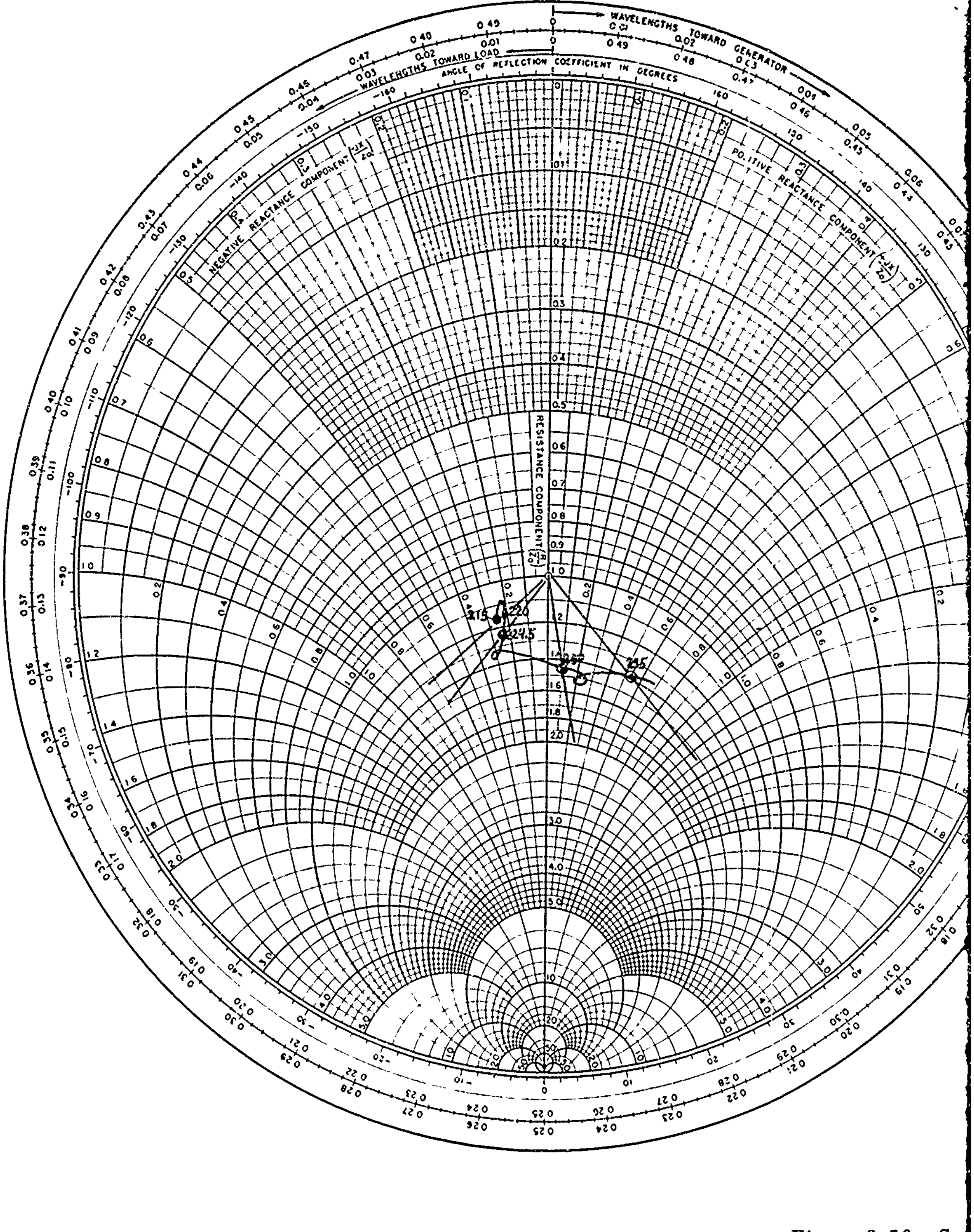


Figure 3-56. Sm
Impedance VS Fr
(After Matching)

A

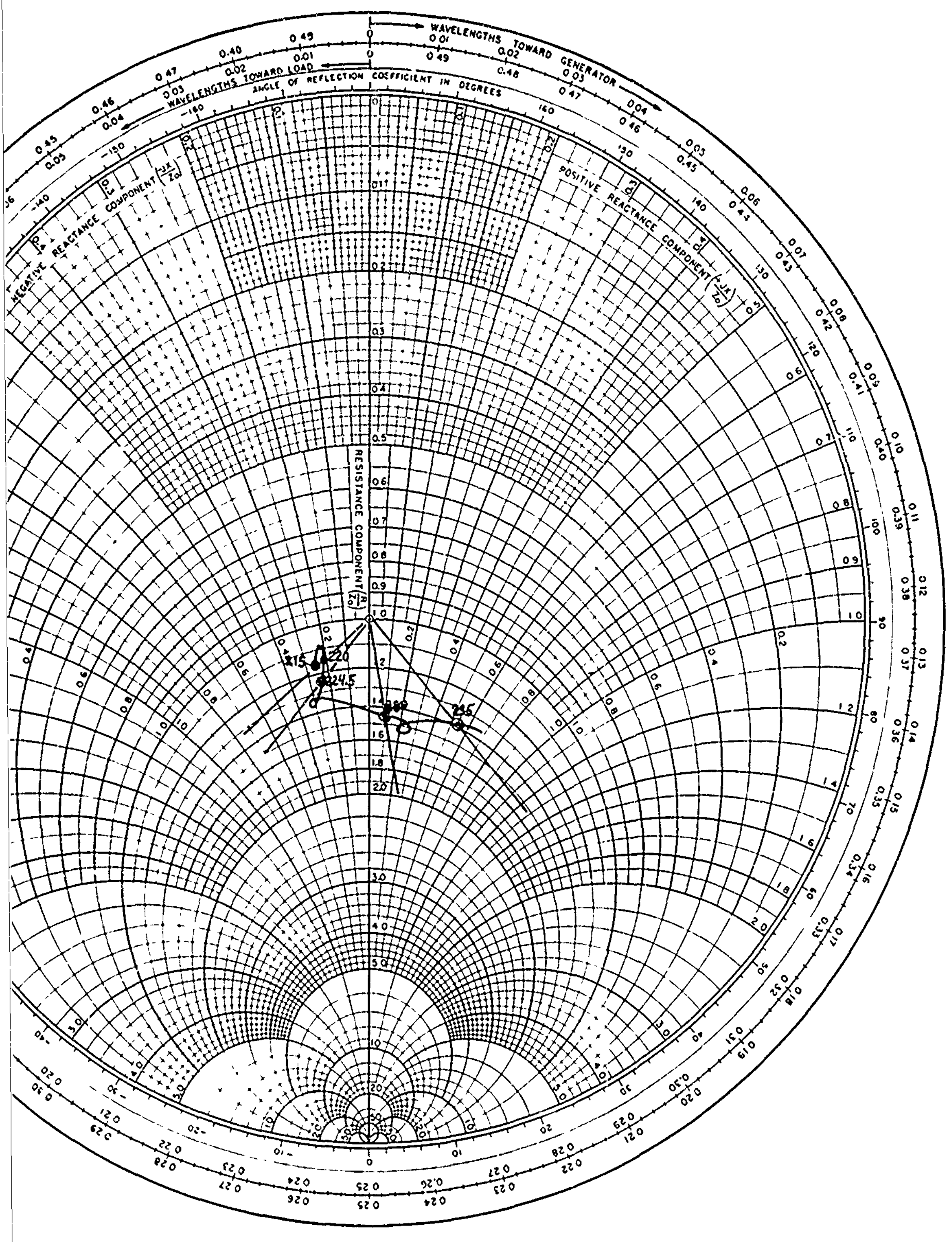


Figure 3-56. Smith Chart of 224.5 MC Antenna Impedance VS Frequency for Spacecraft No. 3 (After Matching)

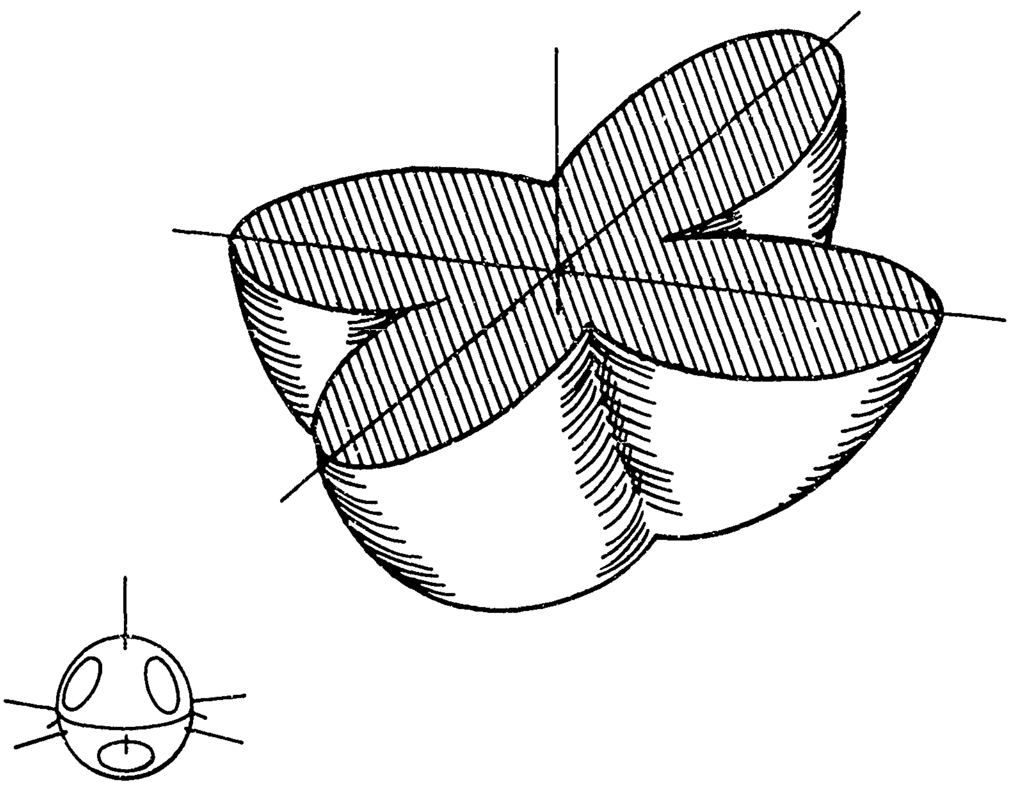


Figure 3-57. Cross Section of 224.5 MC Antenna Radiation Pattern

utilized to measure the impedance, Figures 3-59 and 3-60 are plots of impedance vs frequency for Spacecraft Nos. 2 and 3 after the matching networks were inserted. Table XII is a list of the data collected during the impedance vs frequency measurement made on Spacecraft No. 3. The data is considered to be typical for all three models.

3.8.4.3.2 This antenna system provides nearly omnidirectional coverage. Figure 3-61 is a pictorial view of the radiation pattern from the antennas. This pattern can best be described as three doughnuts intersecting at 60° with respect to each other. The basic element of the antenna, the dipole, normally radiates no energy in the cross-polarization plane. However, as the shell of the Spacecraft is large with respect to a wavelength, the ideal antenna pattern is contaminated, and the shell provides some radiation. This energy fills in the nulls in the antenna patterns so that the deepest minima for any polarization at any aspect angle is less than -36 db. Figure 3-62 and 3-63 shows plot of both horizontal and vertical polarization indicating the improvement that could be achieved by using circular polarization for the ground antenna systems.

3.8.5 Fabrication of Antennas. Cubic constructed the antennas utilized on all three Spacecraft. The following paragraphs describe the major mechanical details of each type of antenna.

3.8.5.1 Fabrication of 136.11 MC Antenna. The assembled 136 mc antenna is shown in figure 3-64. The antenna is attached to the Spacecraft at the top of the battery compartment. The antenna protrudes through an opening in the top plate of the Spacecraft outer shell. The shell is secured around the opening by means of screws into the mount.

3.8.5.1.1 The mechanical assembly is shown in the exploded view in figure 3-65. The insulator block is retained and fastened to the Spacecraft by means of a cup. The lower part of the antenna is a threaded bolt which passes through the insulator block and is fastened by means of a lock washer and nut. The rf connector passes through the side wall of the cup and insulator and is attached to the bottom of the threaded section of the antenna. The insulator block provides the mechanical strength for the antenna. The upper end of the lower half of the antenna is tapered to mate with the top half of the antenna. The bottom of the upper half of the antenna

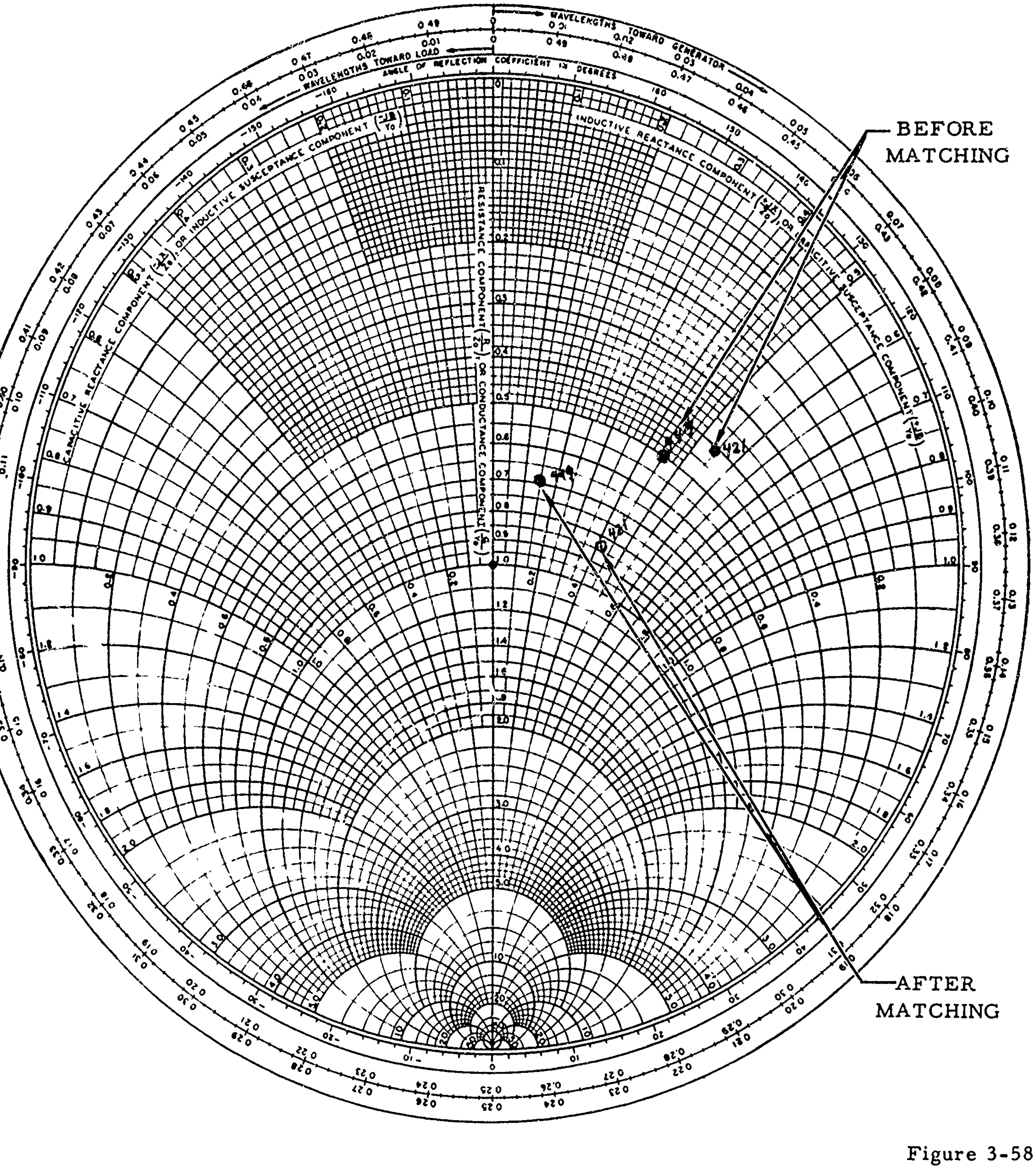


Figure 3-58. Sr
421-449 MC Ant
for Spacecraft N

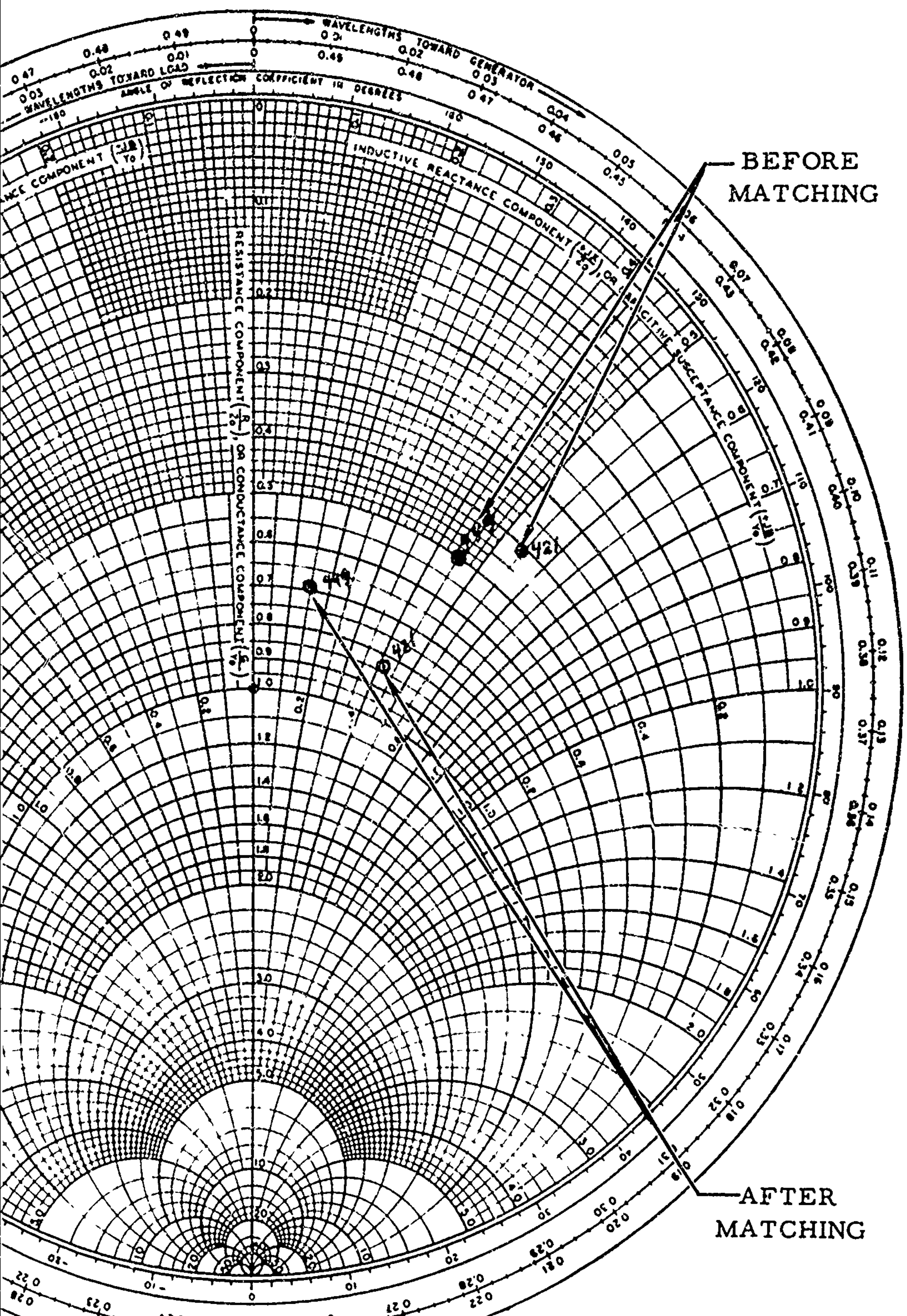


Figure 3-58. Smith Chart of 421-449 MC Antenna Impedance for Spacecraft No. 1

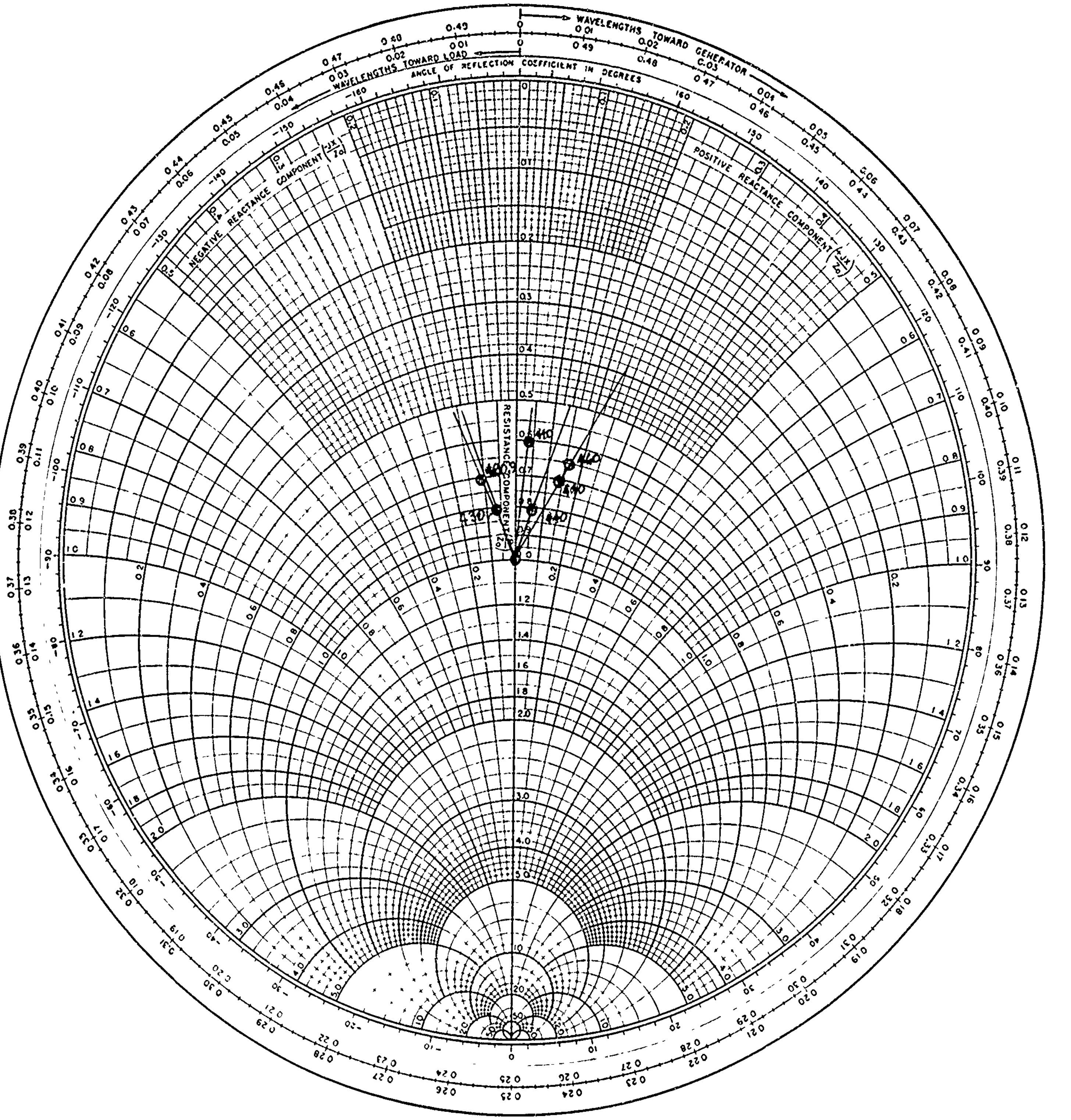


Figure 3-59. Smith Chart 421-
Antenna Impedance VS Frequen
Spacecraft No. 2 (After Matchi

A

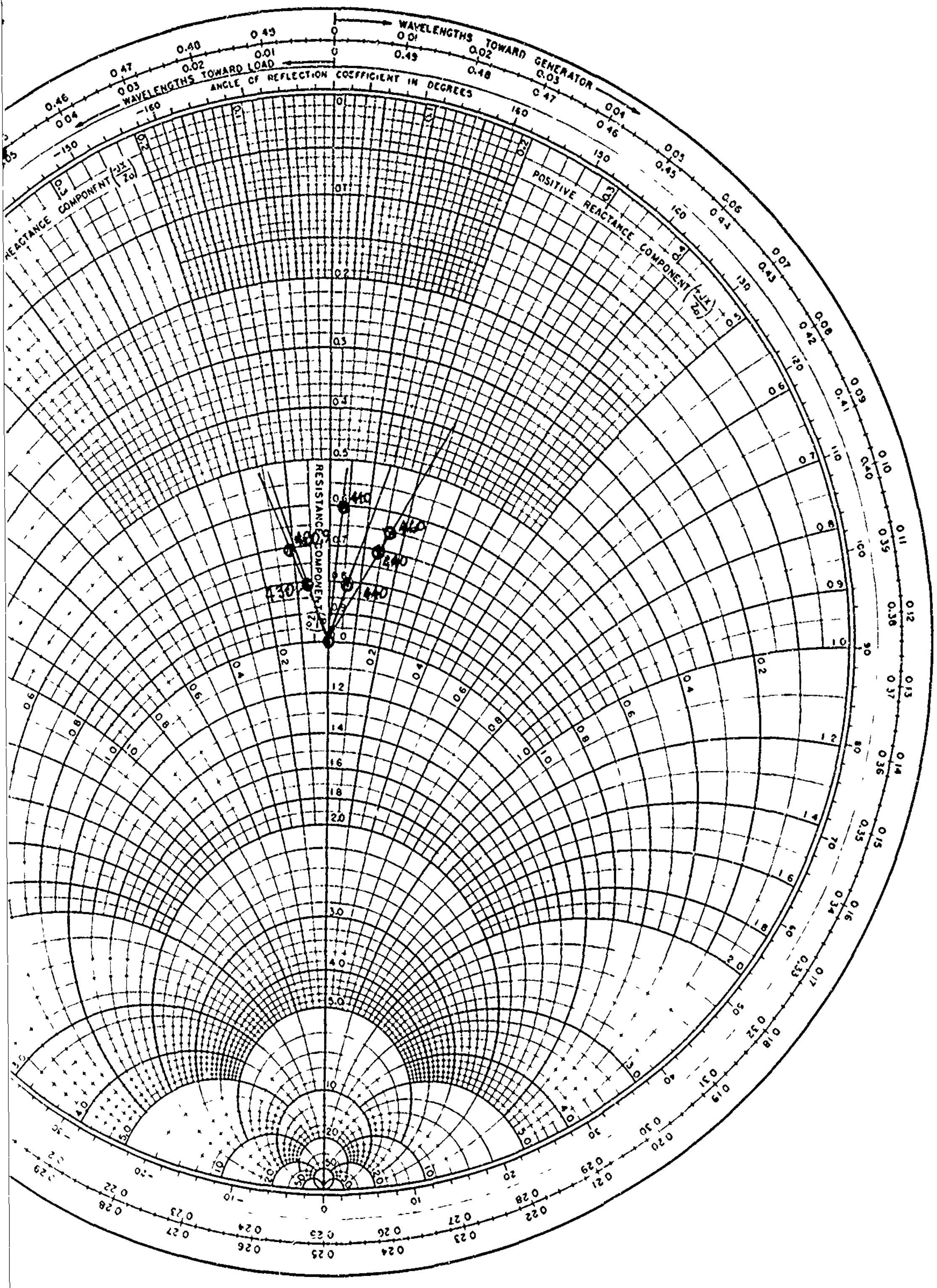


Figure 3-59. Smith Chart 421-449 MC Antenna Impedance VS Frequency for Spacecraft No. 2 (After Matching)

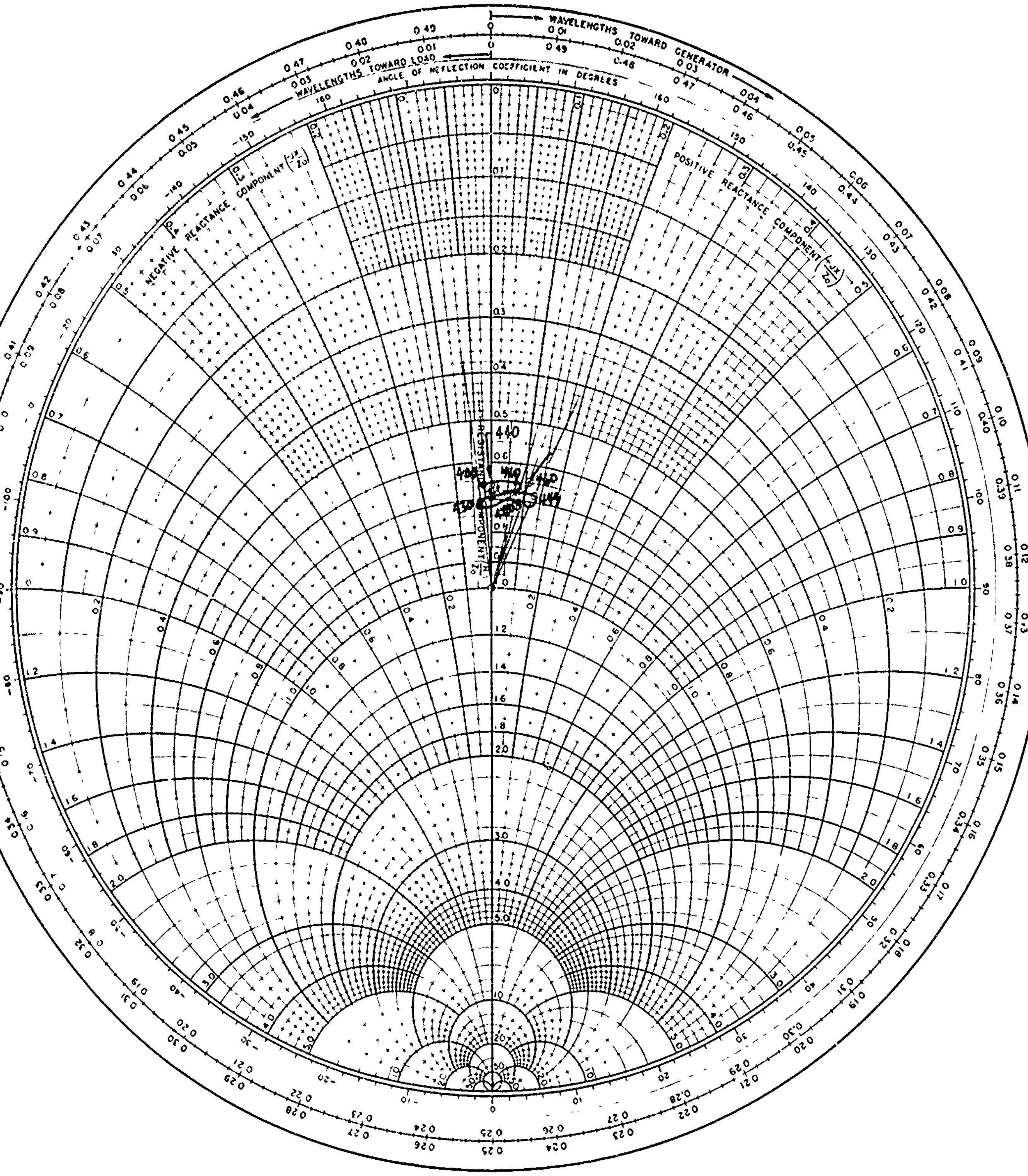


Figure 3-60. Smi
421-449 MC Anten
VS Frequency for
No. 3 (After Ma

A

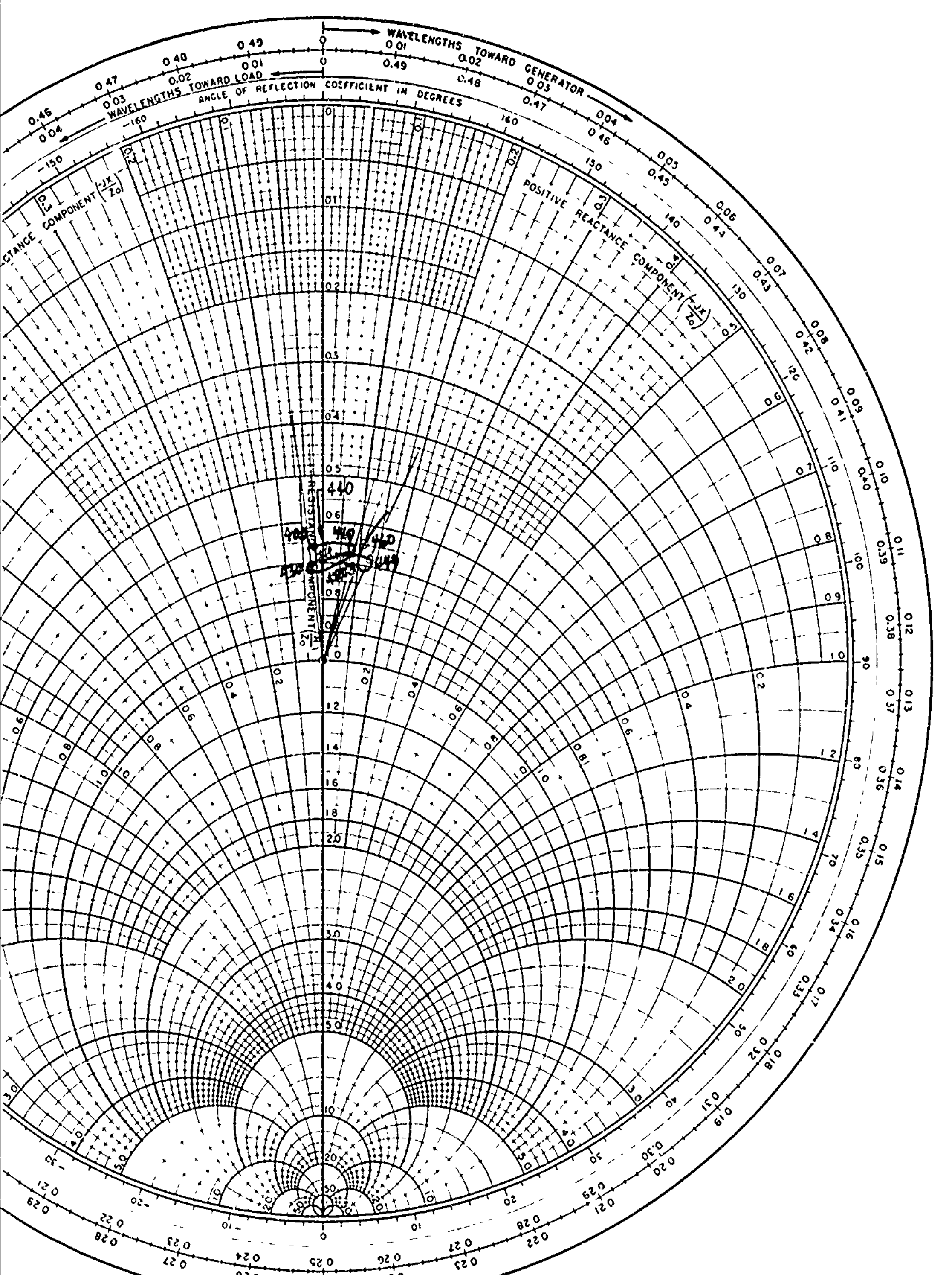


Figure 3-60. Smith Chart of 421-449 MC Antenna Impedance VS Frequency for Spacecraft No. 3 (After Matching)

B

TABLE XII

IMPEDANCE VS FREQUENCY FOR 421-449 MC ANTENNA ARRAY (SPACECRAFT NO. 3)

Frequency	VSWR	Angle of Reflection Coefficient	
		Reference - Short	Load
400 MC	1.50	$+92^{\circ}$	$+87^{\circ}$
410	1.49	$+44^{\circ}$	$+60^{\circ}$
420.9	1.45	-14°	0°
430	1.40	-65°	-73°
440	1.42	-109°	-109°
449	1.50	-160°	-135°
460	1.57	$+149^{\circ}$	$+169^{\circ}$

is machined to a matching taper to fit tightly over the tapered section of the lower half of the antenna. A spring inside the upper tube maintains pressure on the joint to insure good electrical contact.

3.8.5.1.2 The antenna is folded by pulling the two sections apart freeing the hinge joint. With the joint free, the top section can be folded to one side or the other. The antenna must be restrained in the folded position because as soon as it is released the spring causes it to erect.

3.8.5.2 Fabrication of 224.5 MC Antennas. The mechanical assembly of one of the 224.5 mc antennas is shown in figure 3-66. The antenna is attached to the Spacecraft by means of an insulated block which slides through the opening in the shell into the recessed cup attached to the internal bracing. The insulator block is retained in the antenna cup by means of two screws.

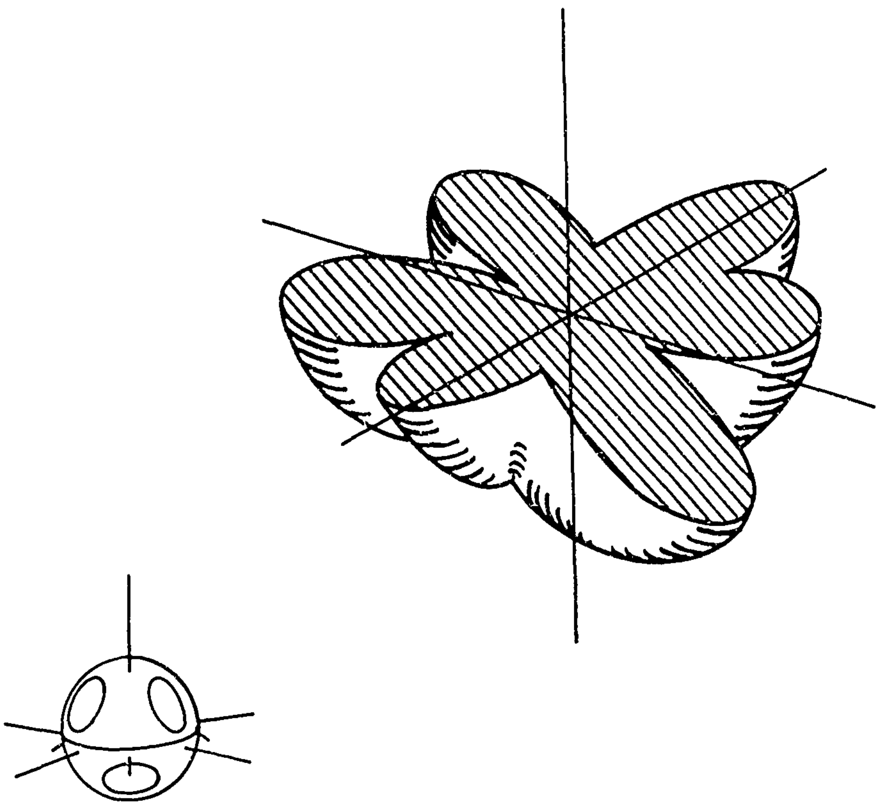


Figure 3-61. Cross Section of 421-449 MC
Antenna Radiation Pattern

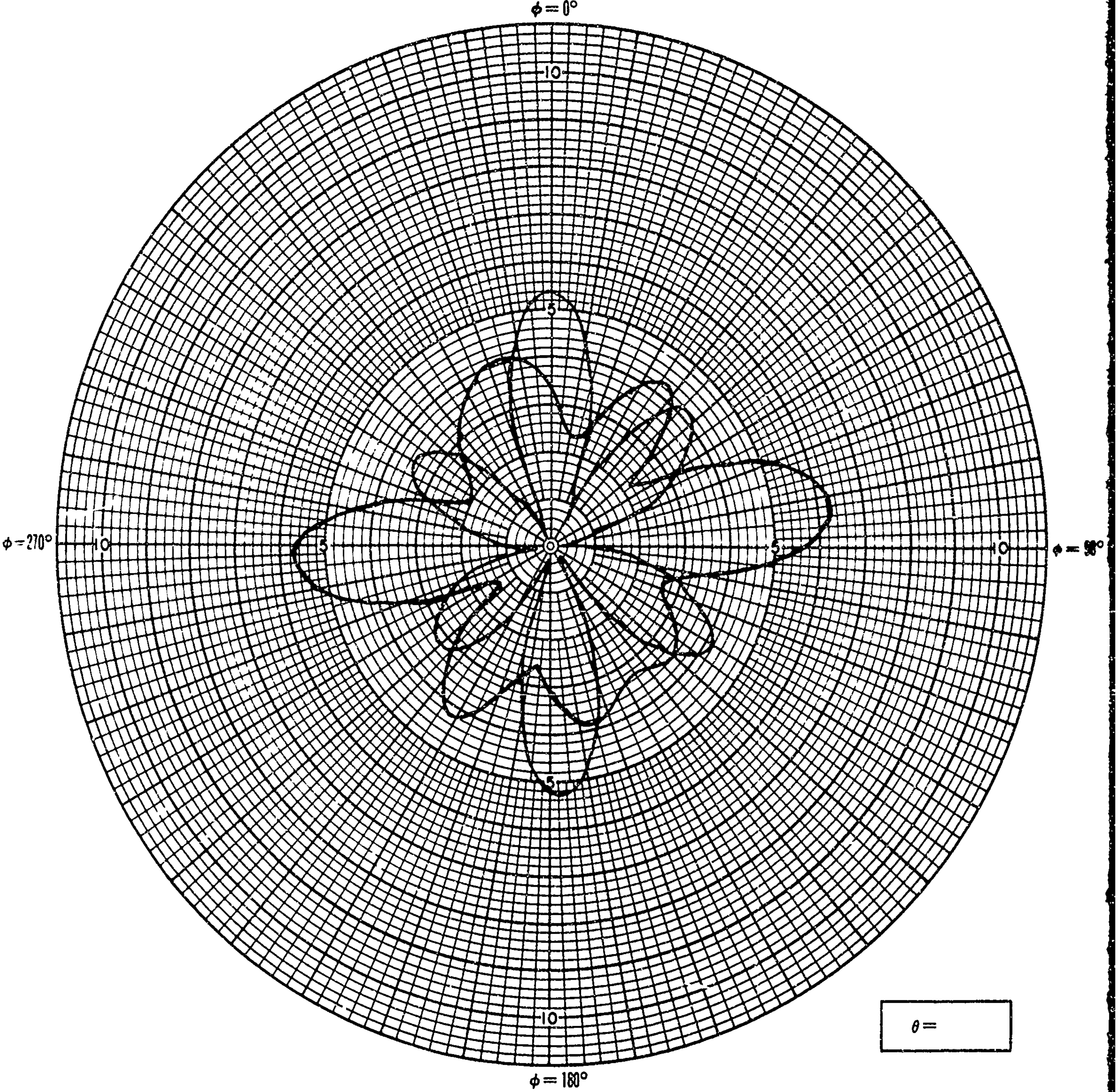


Figure 3-62. Radiation
Horizontal and Vertical
(421 MC)

ñ

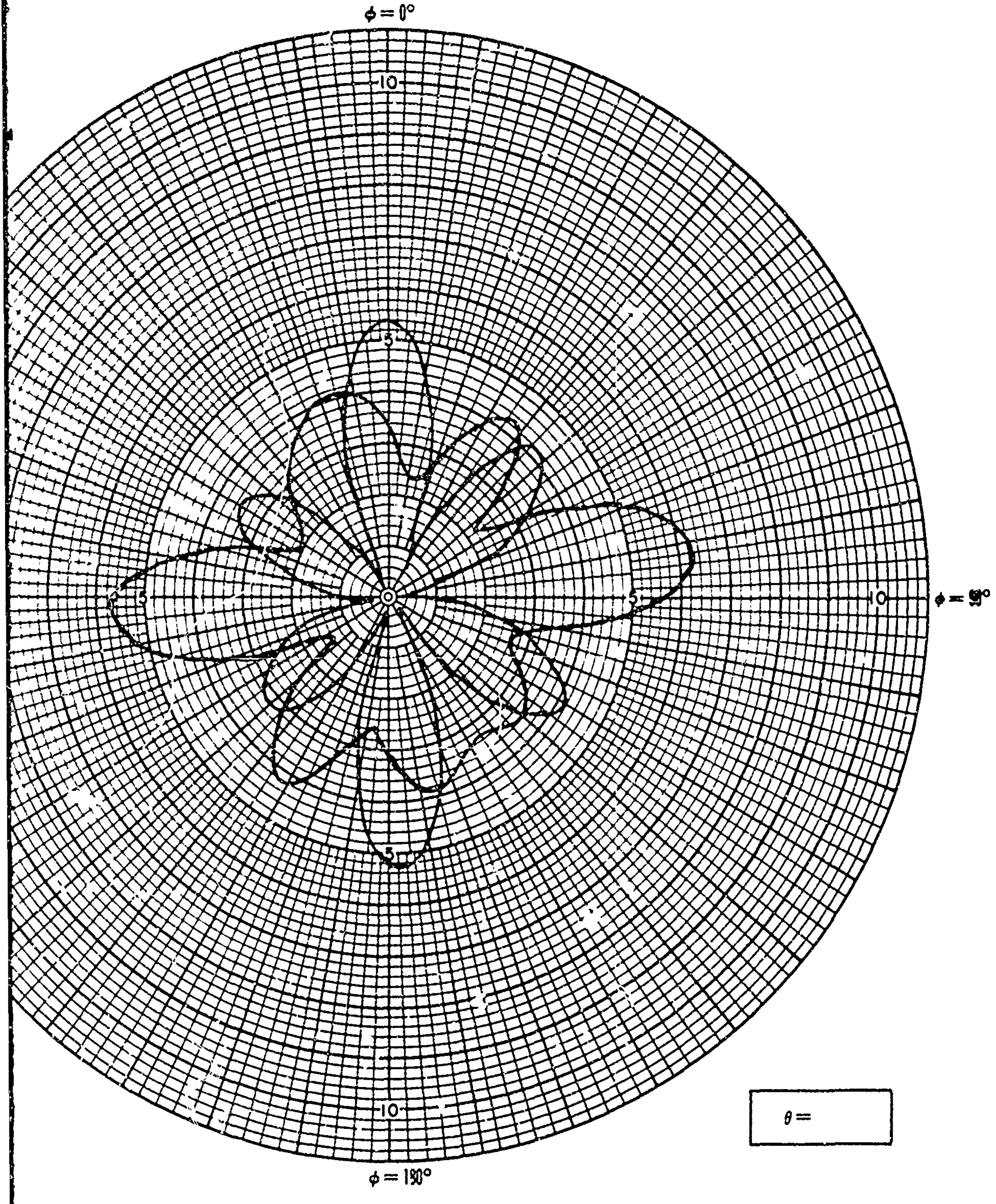


Figure 3-62. Radiation Pattern for Horizontal and Vertical Polarization (421 MC)

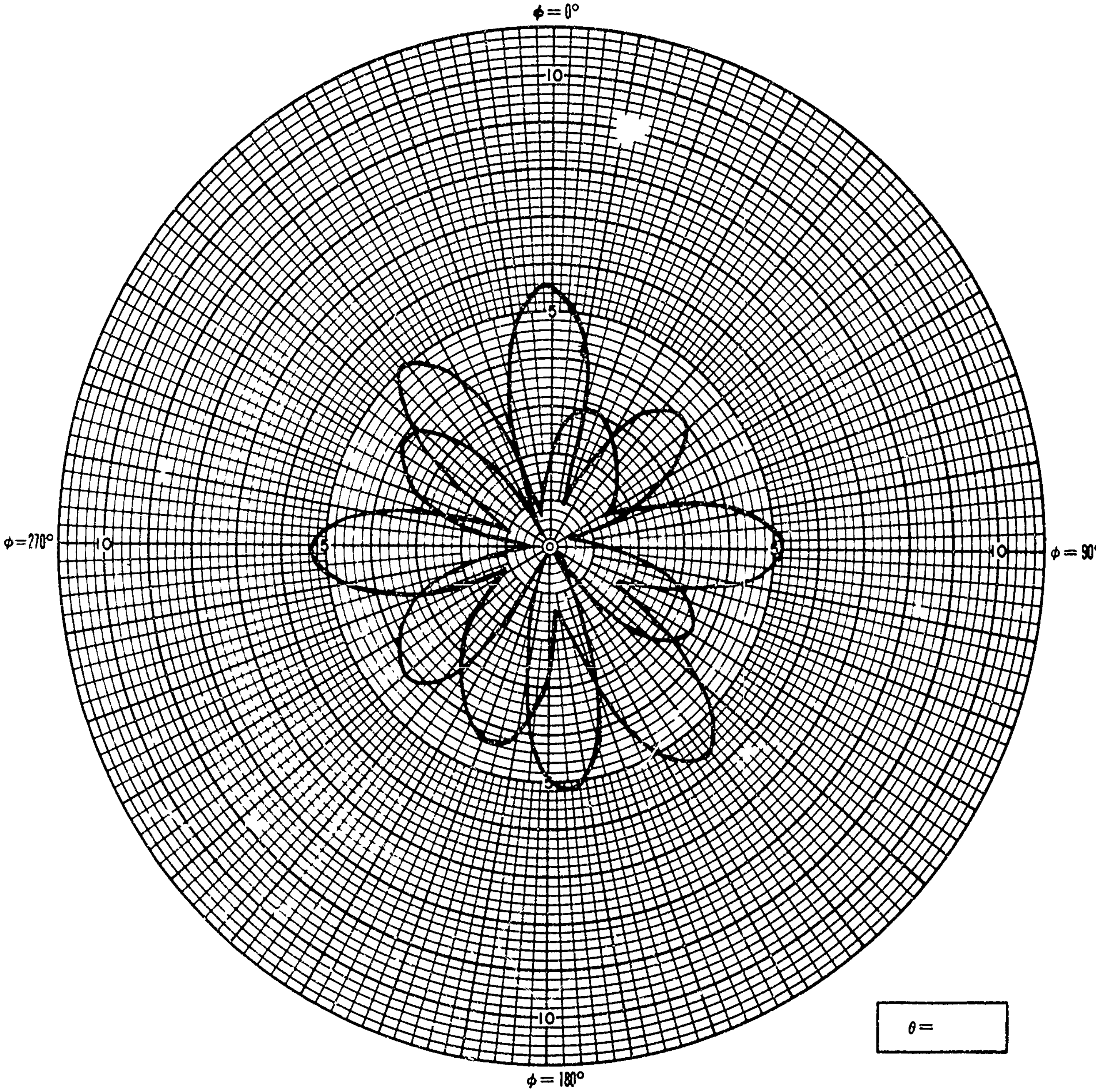


Figure 3-63. Radiation Patterr
Horizontal and Vertical Polariz
(449 MC)

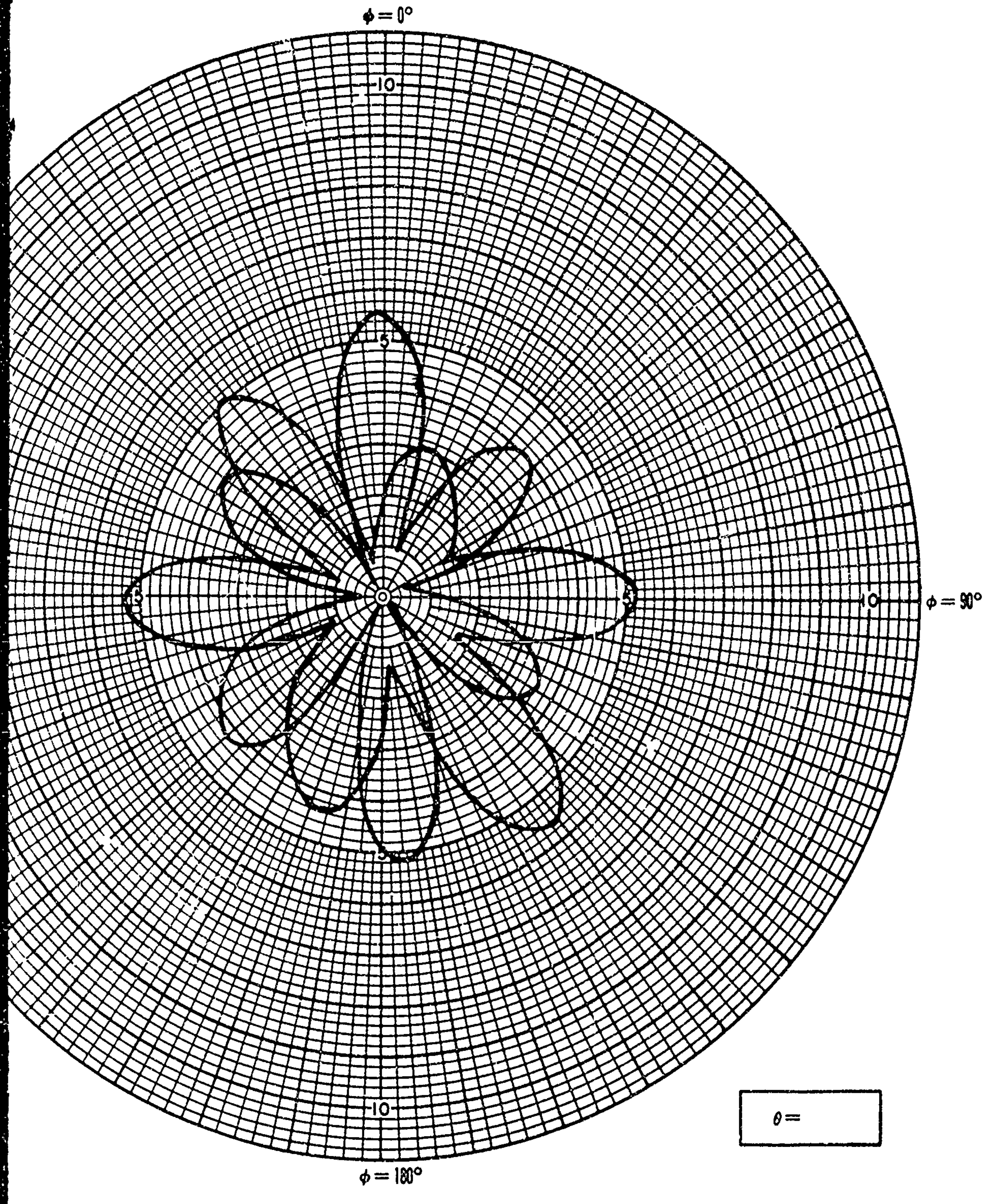


Figure 3-63. Radiation Pattern for Horizontal and Vertical Polarization (449 MC)

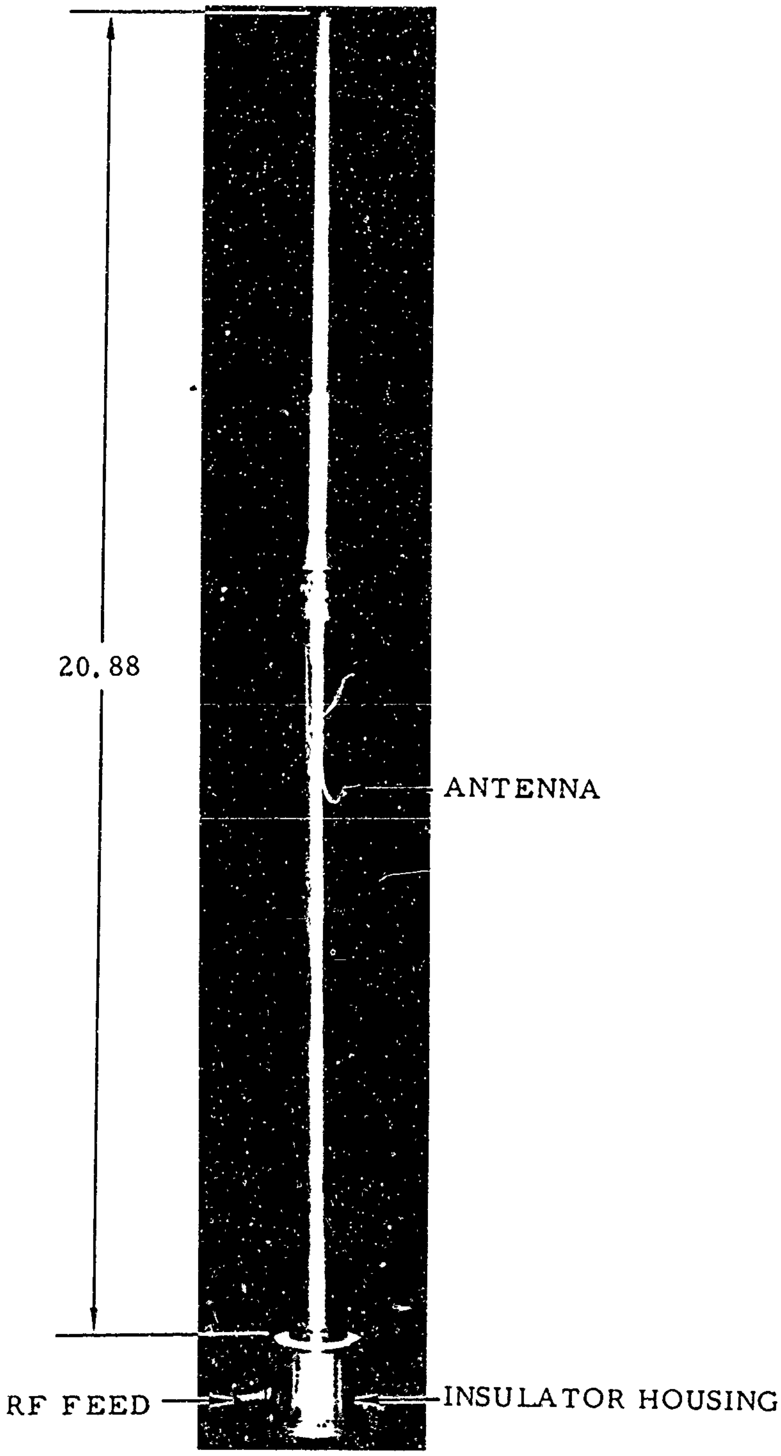


Figure 3-64. Typical Assembled 136.11 MC Antenna

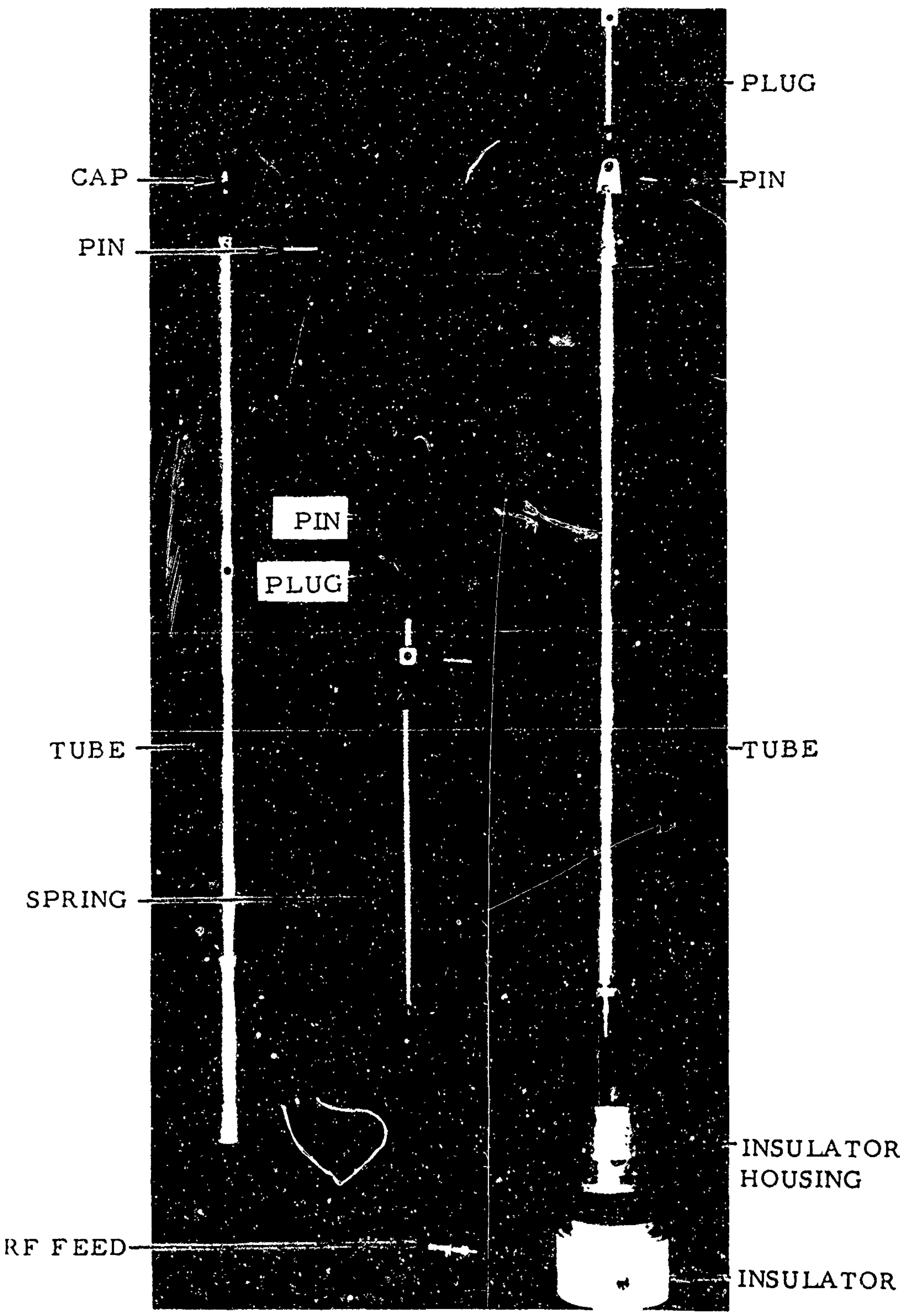


Figure 3-65. Typical Disassembled 136.11 MC Antenna

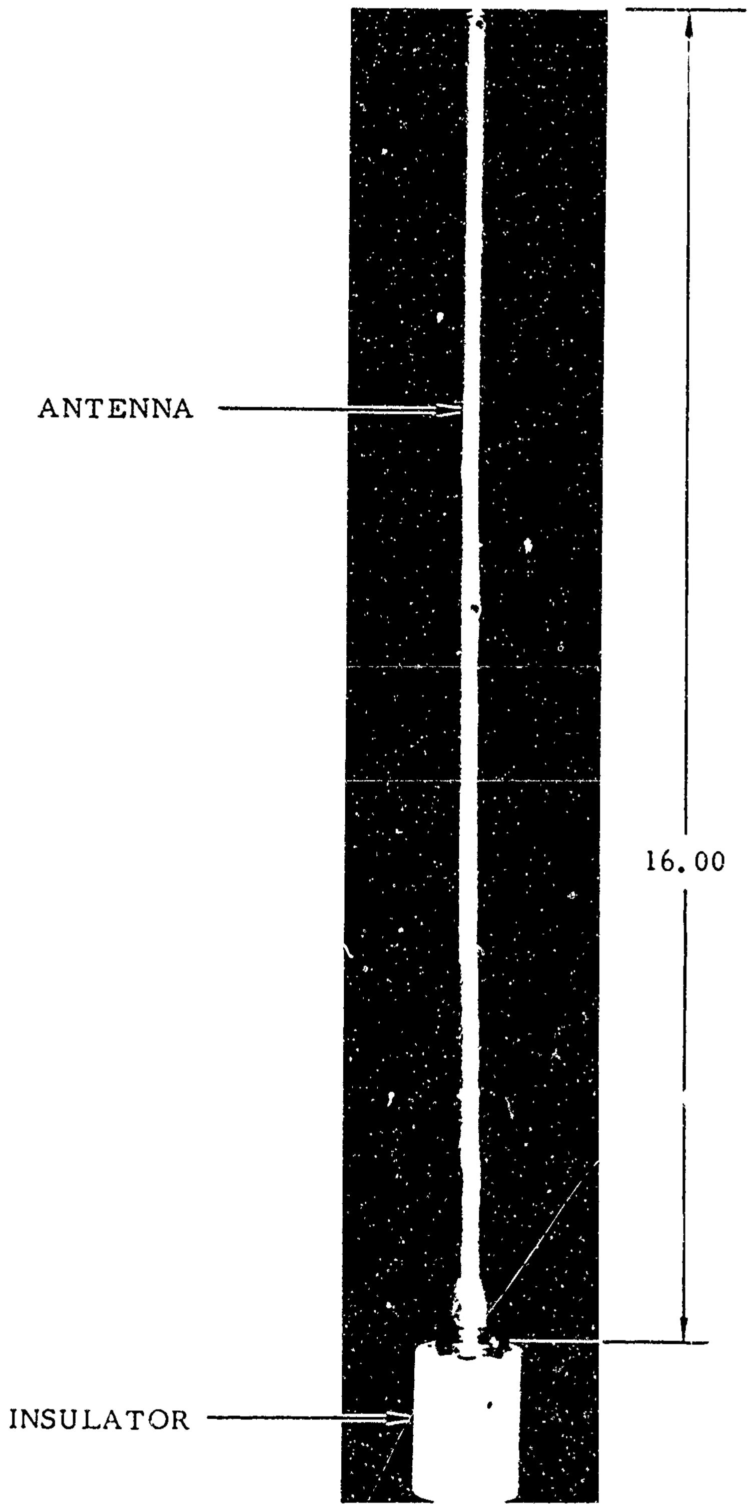


Figure 3-66. Typical Assembled 224.5 MC Antenna

3.8.5.2.1 The mechanical assembly is shown in the exploded view in figure 3-67. The base of the antenna is a threaded bolt which passes through the insulator block and is fastened by means of a lock washer and nut. The rf connector passes through the side wall of the cup and is attached to the rear of the threaded section of the antenna by means of a screw. (See figure 3-68.) The insulator block provides the mechanical strength for the antenna. The outer end of the shank is tapered to mate with the antenna. The antenna is machined to a matching taper to fit tightly over the tapered section of the shank. A spring inside the antenna tube maintains pressure on the joint to insure good electrical contact.

3.8.5.2.2 The antenna is folded by pulling the two sections apart freeing the hinge joint. With the joint free, the antenna can be folded up or down. The antenna must be restrained in the folded position because as soon as it is released the spring causes it to erect. The antenna immediately adjacent to the flight plug has been modified to fold at an angle 76.5 ~~15~~ degrees above the equator of the Spacecraft. This change was required in order to make the spacecraft compatible with the Composite I Multiple Satellite launch configuration.

3.8.5.3 Fabrication of 421-449 MC Antennas. The mechanical assembly of one of the 421-449 mc antennas is shown in figure 3-69. The antenna is attached to the Spacecraft by means of an insulated block which slides through the opening in the shell into the recessed cup attached to the internal bracing. The insulator block is retained in the antenna cup by means of two screws.

3.8.5.3.1 The mechanical assembly is shown in the exploded view in figure 3-70. The base of the antenna is a threaded bolt which passes through the insulator block and is fastened by means of a lock washer and nut. The rf connector passes through the side wall of the cup and is attached to the rear of the threaded section of the antenna by means of a screw. (See figure 3-68.) The insulator block provides the mechanical strength for the antenna. The outer end of the shank is tapered to mate with the antenna. The antenna is machined to a matching taper to fit tightly over the tapered section of the shank. A spring inside the antenna tube maintains pressure on the joint to insure good electrical contact.

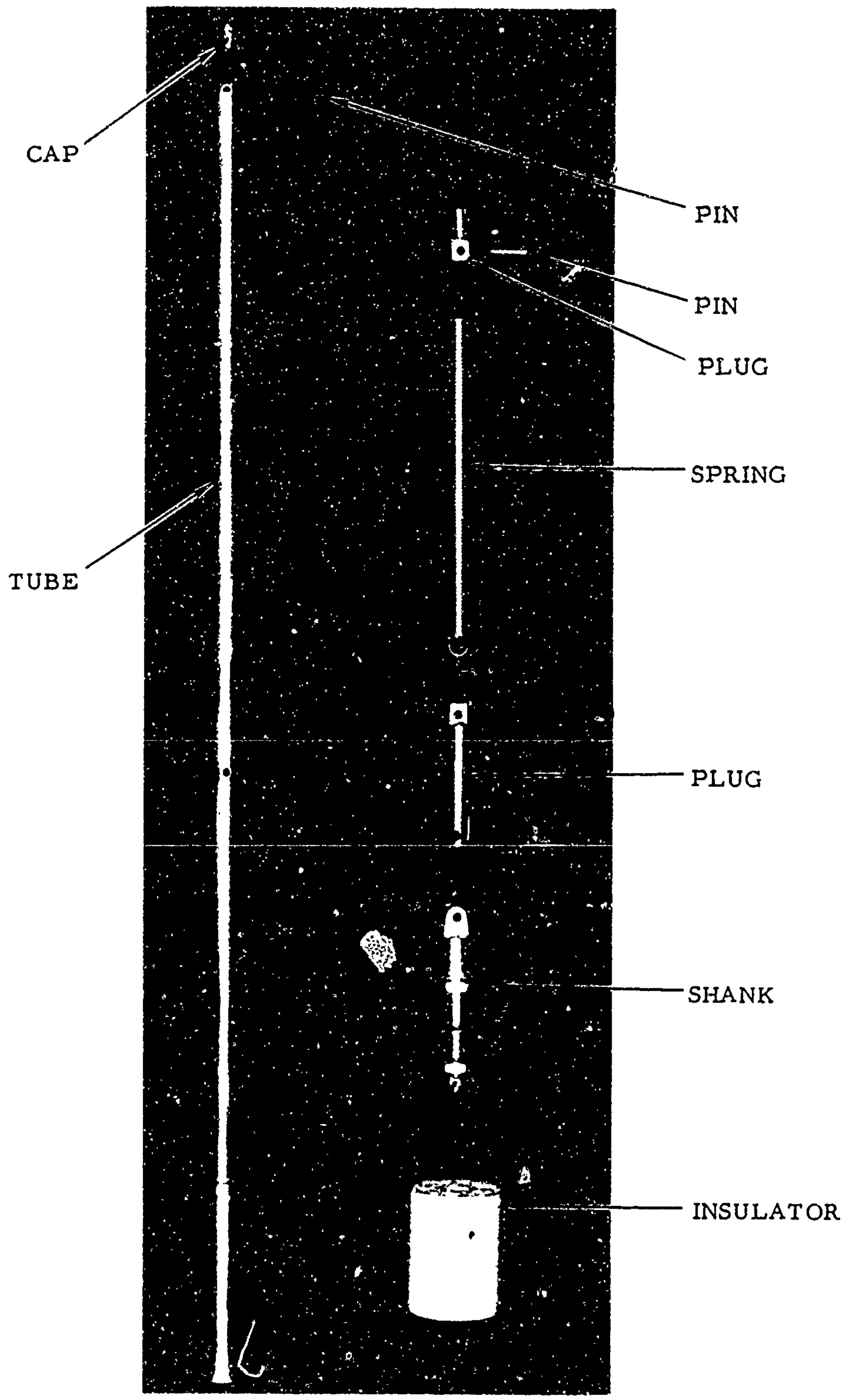


Figure 3-67. Typical Disassembled 224.5 MC Antenna

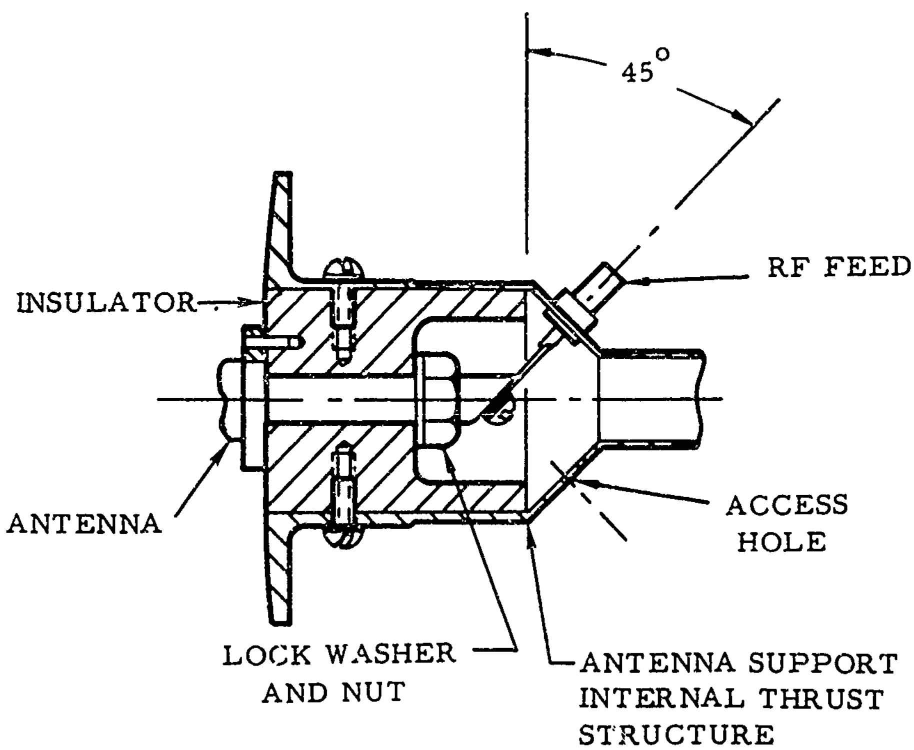


Figure 3-68. Typical Antenna Mounting and RF Attachment

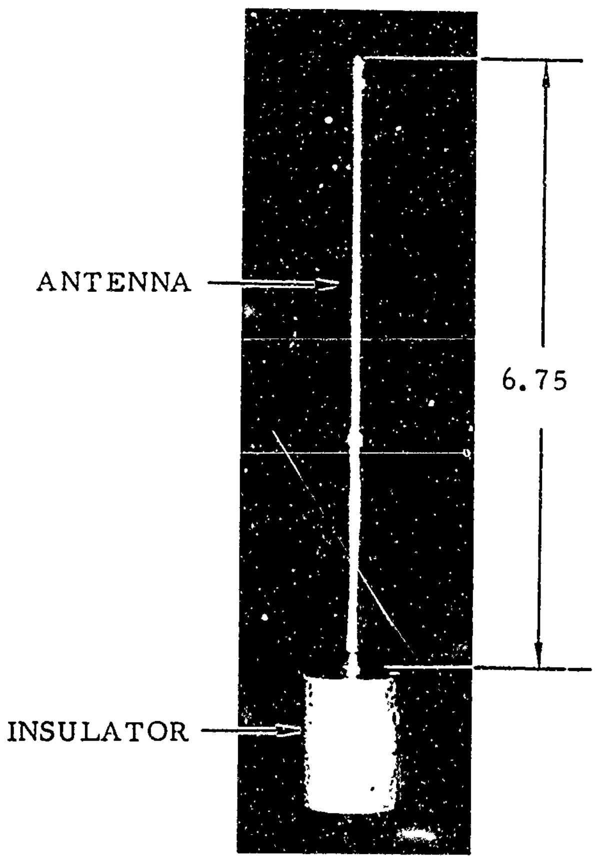


Figure 3-69. Typical Assembled 421-449 MC Antenna

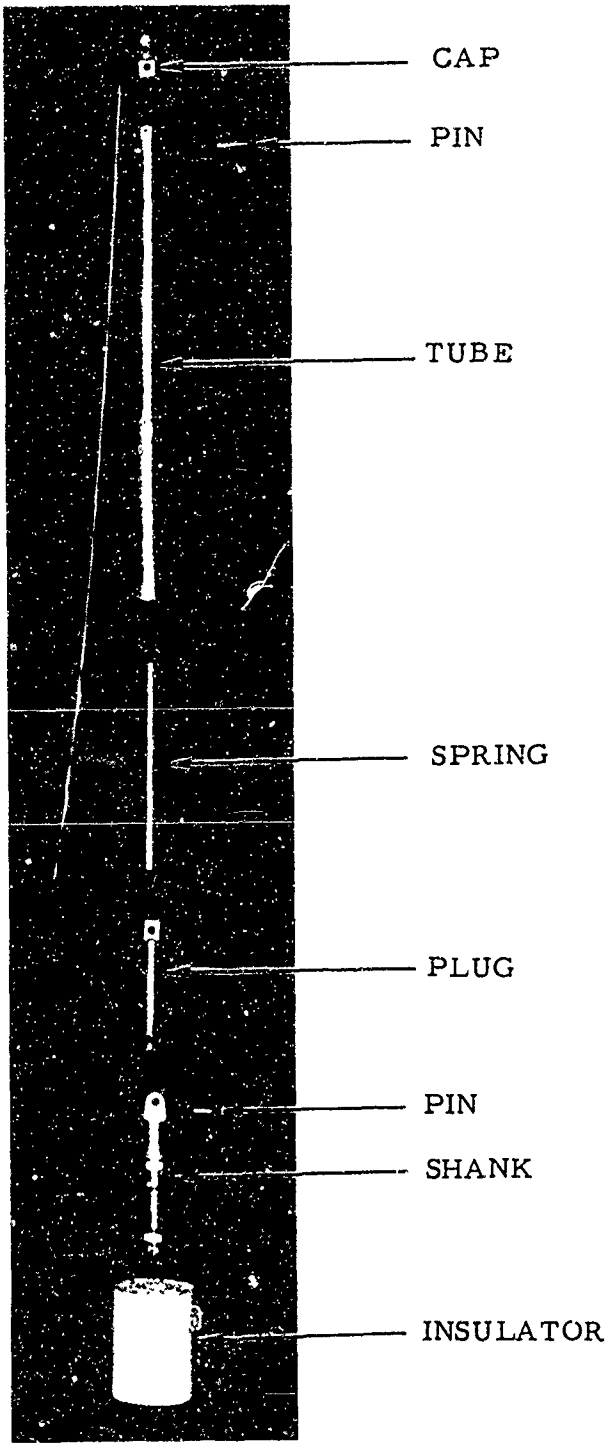


Figure 3-70. Typical Disassembled 421-449 MC Antenna

3.8.5.3.2 The antenna is folded by pulling the two sections apart freeing the hinge joint. With the joint free, the antenna can be folded up or down. The antenna must be restrained in the folded position because as soon as it is released the spring causes it to erect.

3.9 Developmental History of Packaging. The instrumentation support tube, permanently attached to the poles of the internal thrust structure, provides the electronic chassis for mounting the SECOR Transponder and the telemetry and serves as a container for the battery package and power converter. Cubic designed, constructed, and installed the tube in the Spacecraft.

3.9.1 General Problem. The general problem in packaging the spacecraft payload consisted of the design, fabrication, and installation of a chassis which would provide adequate support for the payload components (with respect to the launch and orbital parameters) without increasing appreciably the over-all weight of the Spacecraft. The chassis had to be centrally located with respect to the internal thrust structure, and provide an adequate pressurized compartment for the storage battery package. In addition, the chassis was required to provide room for the electronic components, and to utilize the available volume to the fullest extent.

3.9.2 Preliminary Considerations. Cubic proposed an instrumentation tube similar to the one employed by Vanguard II. The tube is sealed at its lower end and secured to the column at the south pole of the internal thrust structure. The top is supported by a doughnut-shaped ring connected to the upper thrust ring of the internal structure. The interior of the tube is the chamber for the battery package and the power converter. The exterior of the tube forms a central mounting surface for the electronics chassis. As the transponder electronics are mounted on five subchassis, Cubic fitted a pentagonal support structure to the exterior of the instrumentation tube for the purpose of attaching the five transponder subchassis. The available working length of the instrumentation tube is approximately fifteen inches, and the length of the transponder chassis is eight inches. As the vertical support tubes of the internal thrust structure taper toward the central tube at the poles of the structure, the working

volume around the tube is constricted in this area. Thus by locating tne transponder chassis approximately three inches from the south pole, approximately four inches of the support tube are available for the telemetry chassis.

3.9.3 Development Details. Prior to finalizing the design, Cubic conducted a detailed stress and harmonic analysis to determine the size and type of material best suited for the structure, and to resolve the method of welding which would provide adequate support for the battery package. As a reliable pressure-seal for the battery chamber was not available, an alternate method of protecting the batteries from the low pressure environment was selected. The entire battery package was potted in silicone rubber compound, and the power converter was mounted on top. See figure 3-71 for the detailed structure.

3.9.4 Fabrication of the Instrumentation Support Tube. Cubic constructed three instrumentation support tubes. Because of the type of separation fixture utilized on the prototype model, the tube fabricated for this Spacecraft differs slightly from the ones constructed for the flight craft. Originally it was decided to use sheet steel for the pentagonal sleeve. However, prior to fabrication, machined aluminum alloy was selected in order to avoid the incipient drum resonances associated with sheet steel structures.

3.9.5 Installation of the Instrumentation Support Tube. The instrumentation support tube is secured to the base of the thrust structure by screws. The top of the tube is supported by the doughnut-shaped ring which secures to the upper thrust ring of the internal structure. Four screws fasten the top of the tube. (See figure 3-72.)

3.9.6 Installation of the Spacecraft Electronics. Figure 3-73 shows the location of the various electronic components as installed on the instrumentation support tube. Figure 3-74 is an intercomponent wiring diagram. As shown in the wiring diagram, all important monitoring points and power connections are brought out through the flight plug. This allows the power consumed and the performance of each subassembly to be monitored during checkout of the Spacecraft. It also permits the batteries to be charged externally. A shorting plug which interconnects the internal wiring

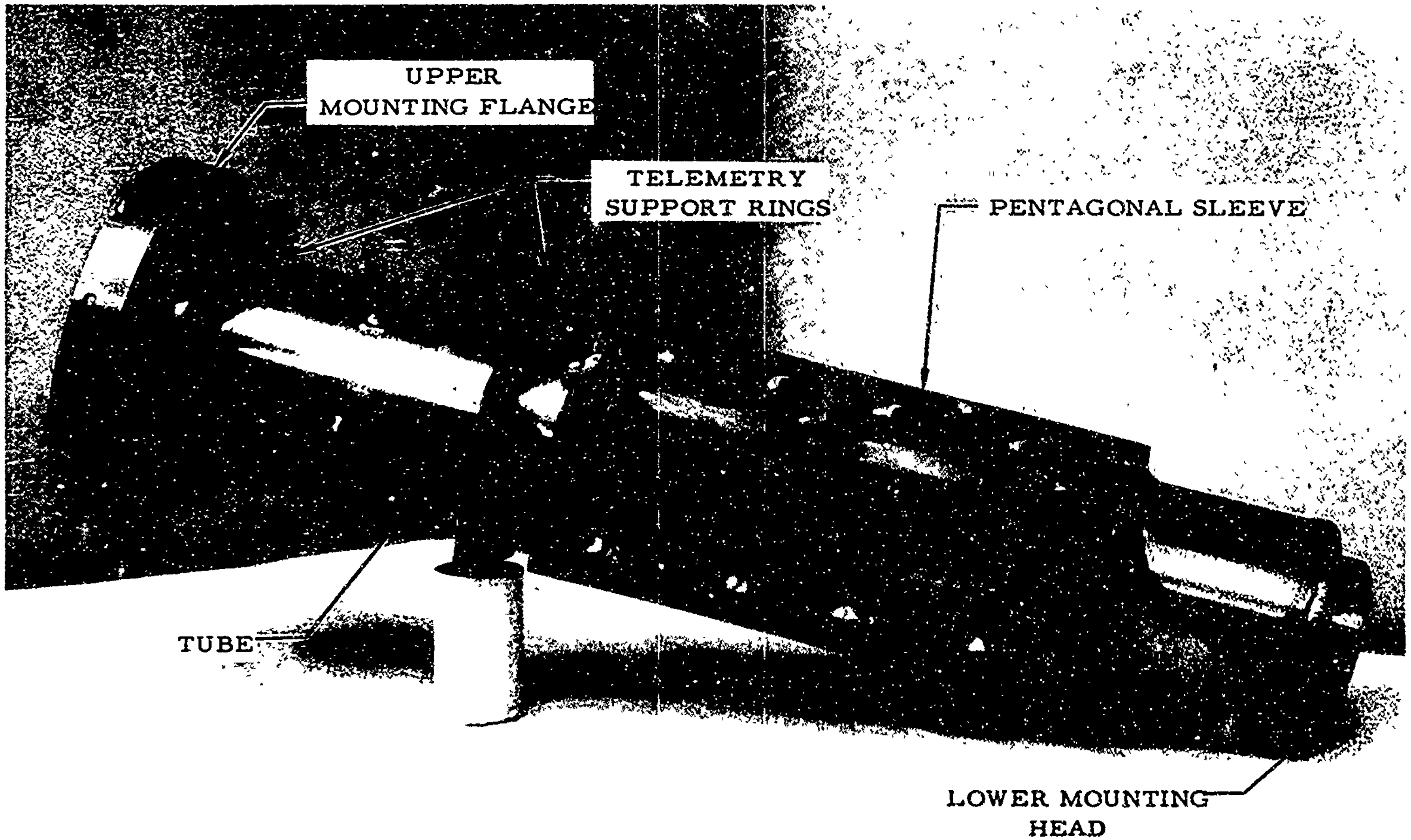


Figure 3-71. Mechanical Details of Instrumentation Support Tube

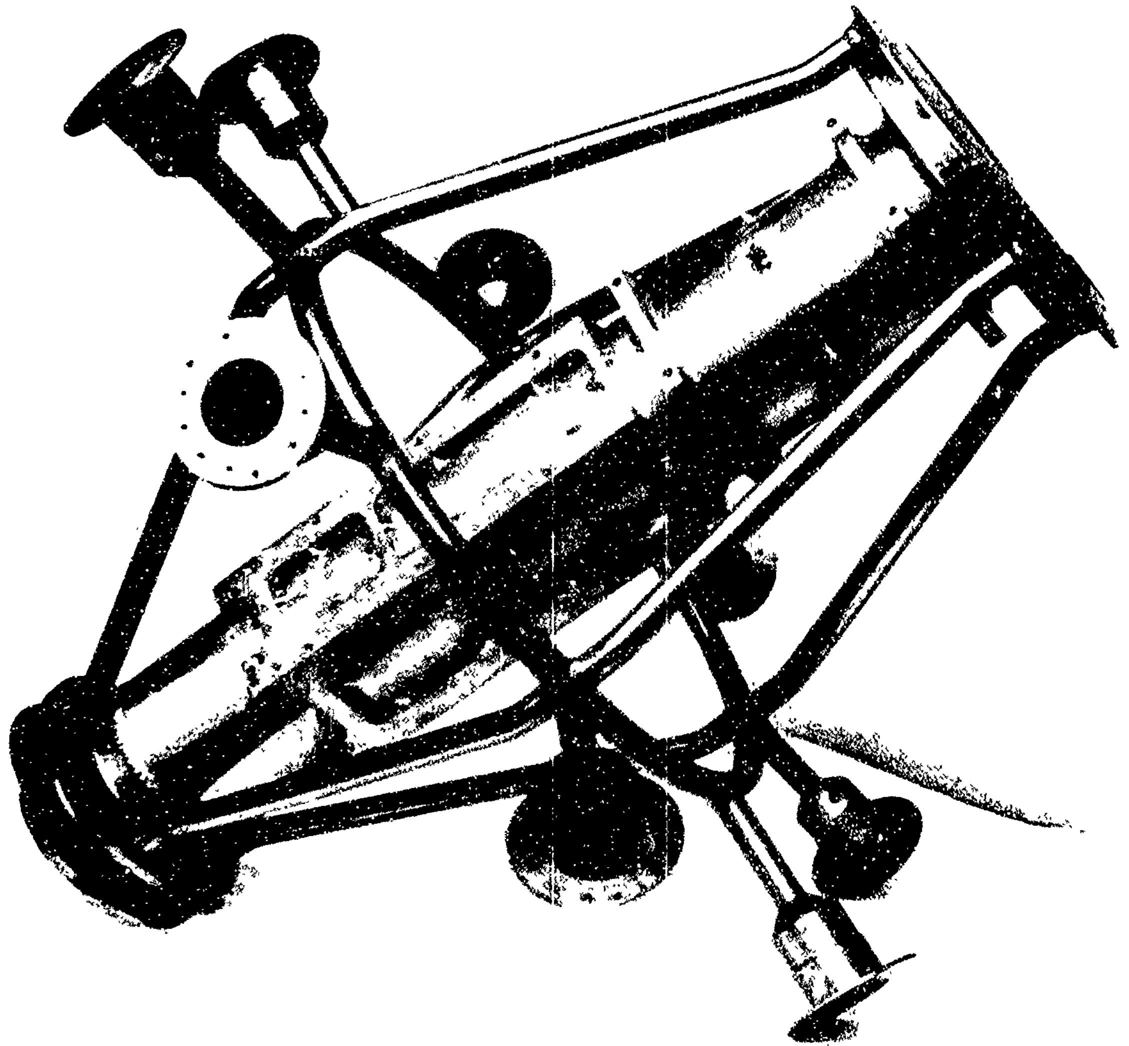
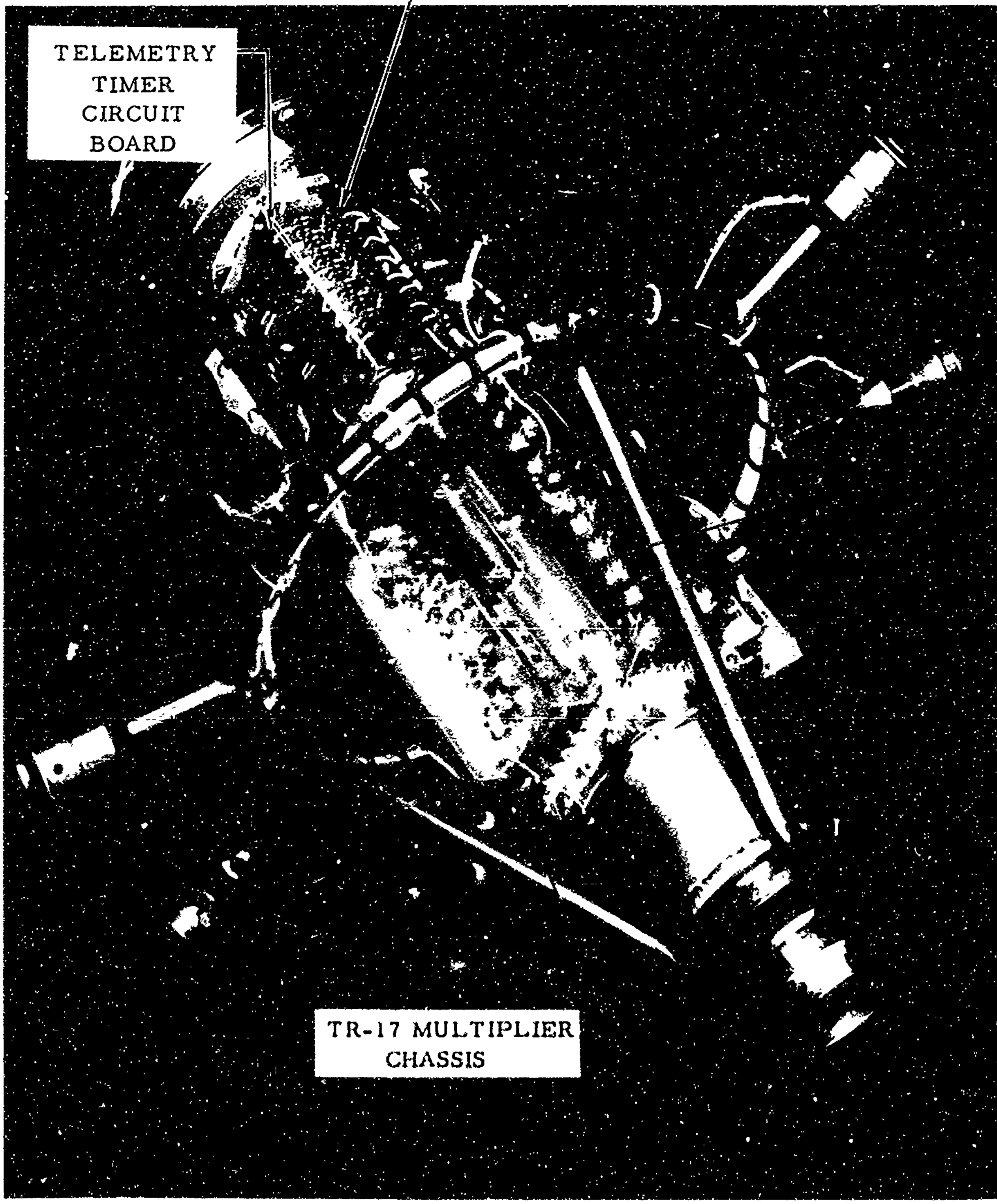
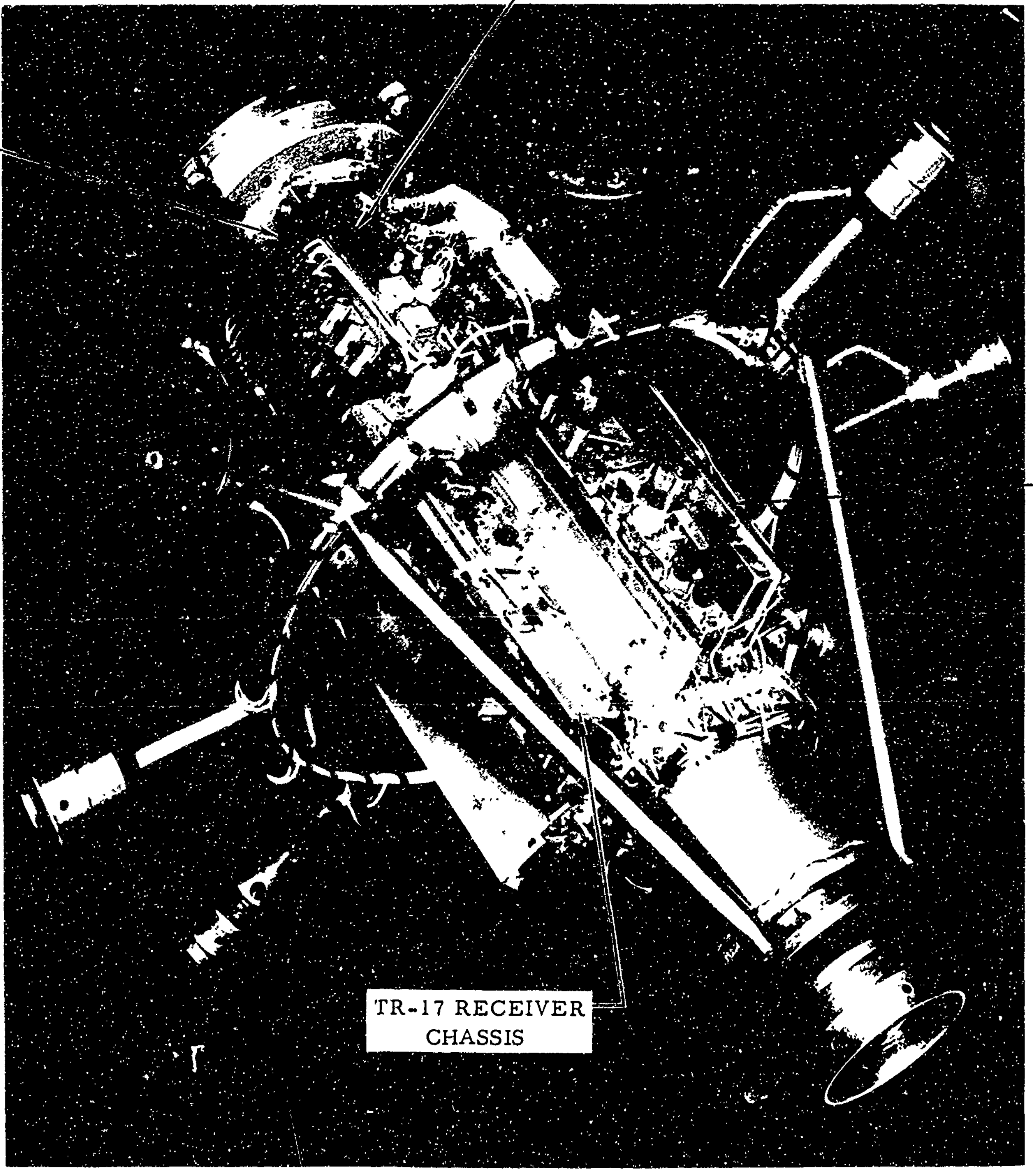


Figure 3-72. Instrumentation Support Tube Installed in Spacecraft No. 1


TELEMETRY COMMUTATOR CIRCUIT BOARD
TELEMETRY TIMER CIRCUIT BOARD
TELEMETRY SENSOR CIRCUIT BOARD
TR-17 COHERENT CARRIER-COR-RELATION DETECIOR CHASSI
TR-17 MULTIPLIER CHASSIS


TELEMETRY
VCO CIRCUIT BOARD
ELEMETRY
SENSOR
CIRCUIT
BOARD
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RIER-COR-
ELATION
CIOR CHASSIS
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A
TR-17 RECEIVER
CHASSIS

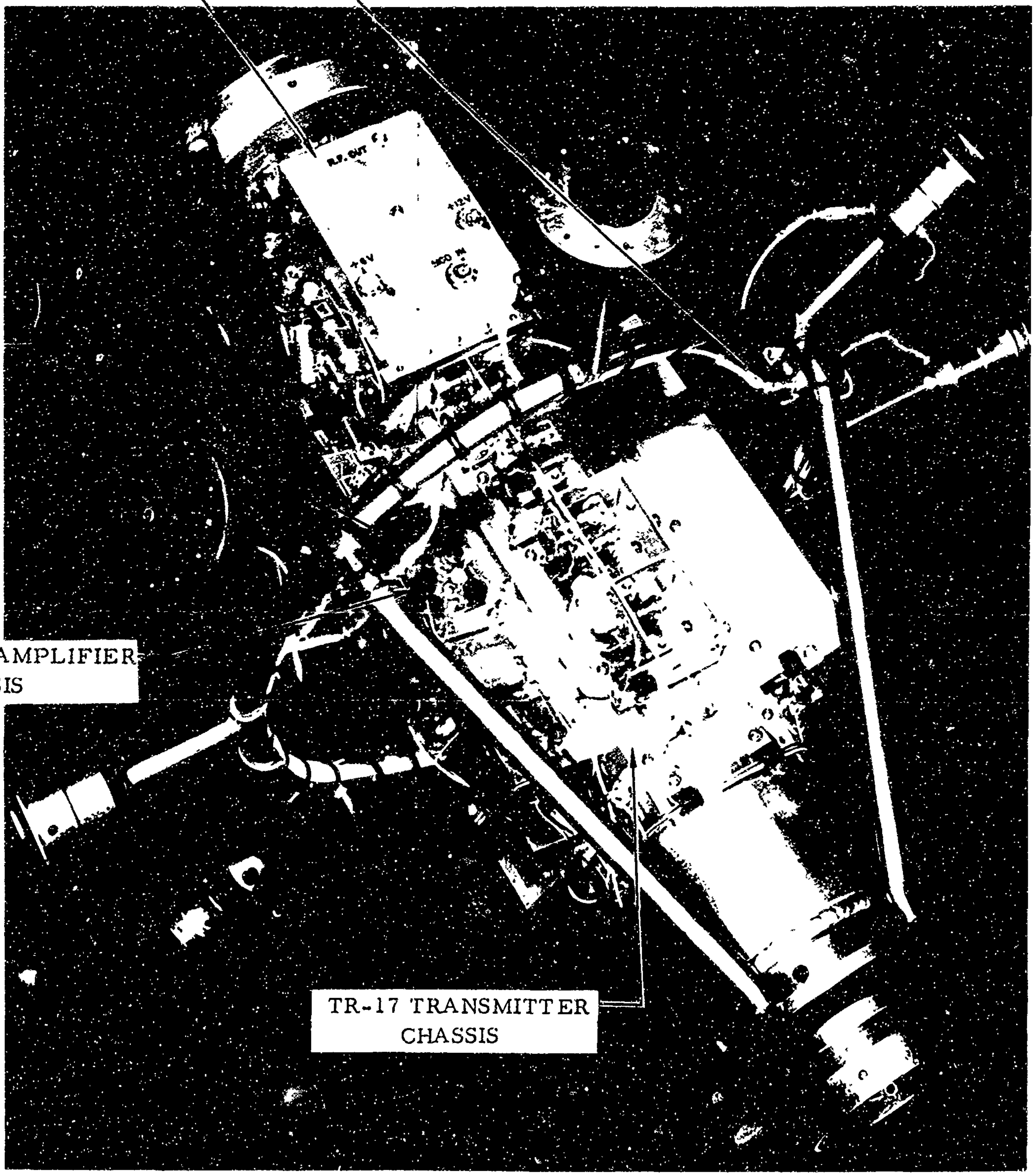
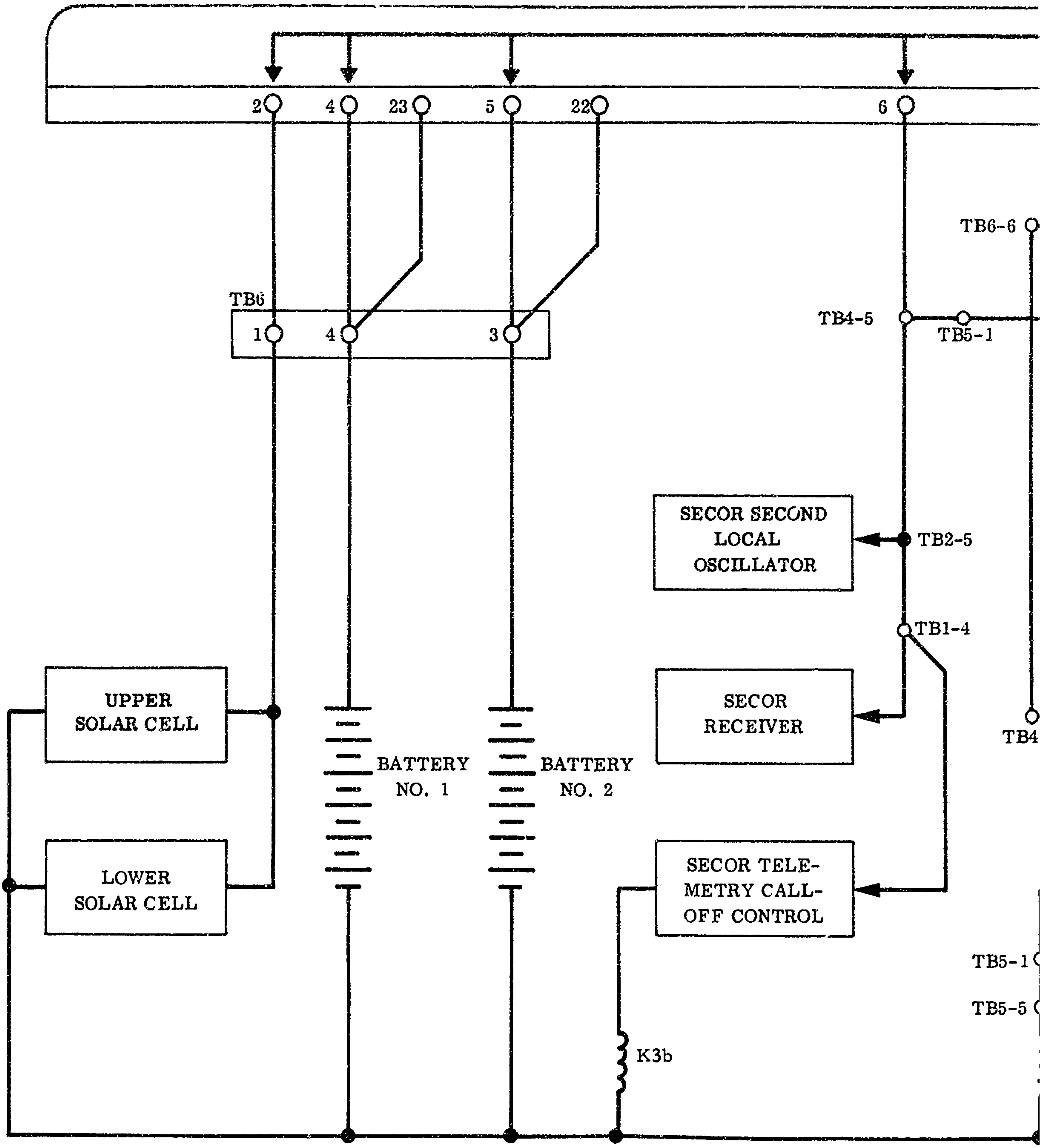
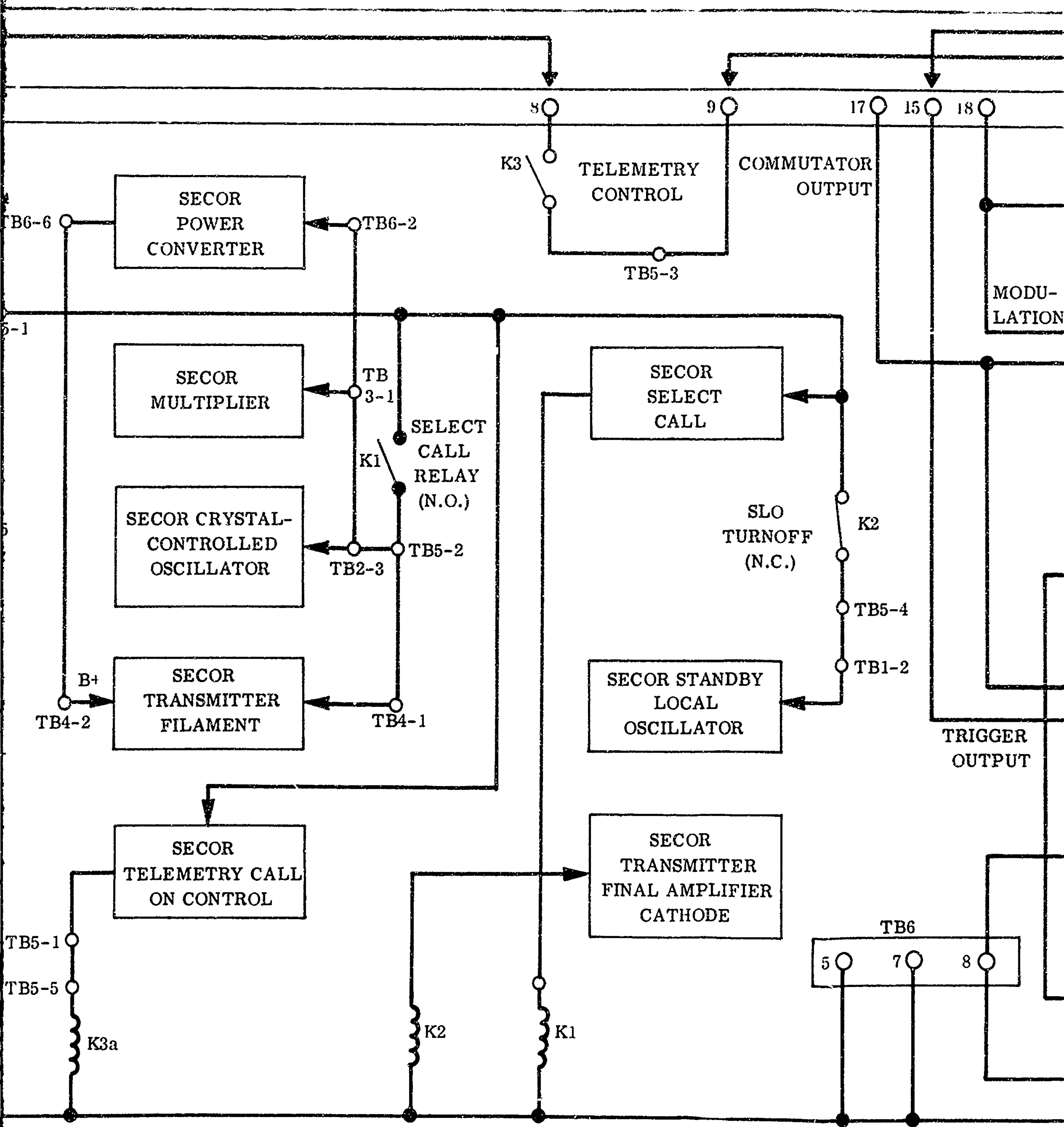


Figure 3-73. Typical Electronic
Component Locations



2
4
23
5
22
6
TB6
1
4
3
TB6-6
TB4-5
TB5-1
SECOR SECOND LOCAL OSCILLATOR
TB2-5
TB1-4
UPPER SOLAR CELL
BATTERY NO. 1
BATTERY NO. 2
SECOR RECEIVER
TB4
LOWER SOLAR CELL
SECOR TELE-METRY CALL-OFF CONTROL
TB5-1
TB5-5
K3b

A



8
9
17
15
18
K3
TELEMETRY CONTROL
COMMUTATOR OUTPUT
TB5-3
SECOR POWER CONVERTER
TB6-6
TB6-2
MODU-LATION
SECOR MULTIPLIER
TB 3-1
SELECT CALL RELAY (N.O.)
K1
SECOR SELECT CALL
SLO TURNOFF (N.C.)
K2
SECOR CRYSTAL-CONTROLLED OSCILLATOR
TB2-3
TB5-2
TB5-4
TB1-2
B+
SECOR TRANSMITTER FILAMENT
TB4-2
TB4-1
SECOR STANDBY LOCAL OSCILLATOR
TRIGGER OUTPUT
SECOR TELEMETRY CALL ON CONTROL
SECOR TRANSMITTER FINAL AMPLIFIER CATHODE
TB6
5
7
8
TB5-1
TB5-5
K3a
K2
K1

B

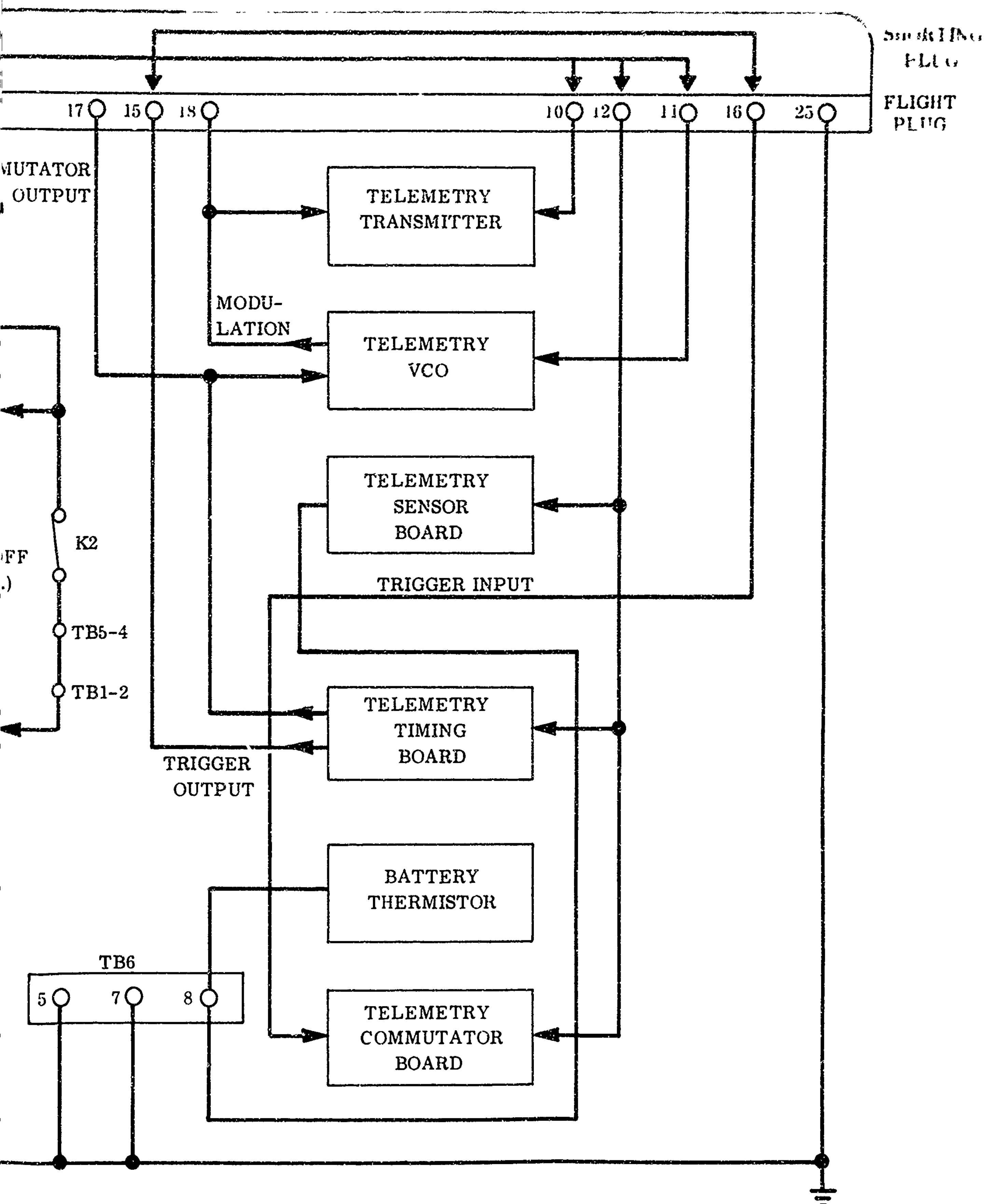


Figure 3-74. System Interconnecting Wiring Diagram

is inserted into the flight plug prior to launch. Figure 3-75 gives an example of the use of the flight plug for monitoring spacecraft performance during a vibration test.

3.10 Developmental History of the Despin Mechanism. In the present satellite-cluster launch configuration, the Geodetic Spacecraft does not require despinning. However, as the originally proposed launch vehicle might have injected the spacecraft into orbit at an excessive spin rate, Cubic investigated various despinning devices.

3.10.1 General Problem. An optional despinning device has been specified to permit spin-reduction after orbit injection. For the SCOUT, the injection spin rate is about 2.7 rps. A nominal spin rate has no adverse effect upon the operation of the complete Geodetic SECOR system. Furthermore, it is definitely undesirable to reduce the spin to zero, as under these conditions one side of the Geodetic Spacecraft will be constantly facing away, thereby establishing an undesirable thermal gradient across ~~acorss~~ the Spacecraft.

3.10.2 Preliminary Considerations. Although some spin retention is desirable, a simple calculation of a "yo-yo" despin device can be made most easily for the case of complete spin stoppage. The actual "yo-yo" will then weigh somewhat less. Assume a 20-inch diameter sphere weighing 30 pounds, initially spinning at 16.7 radians/sec, with a radius of gyration assumed at 0.33 ft; stoppage can be effected in about 0.25 seconds by the use of two 2-ounce weights (each with a 40-inch cable) which are jettisoned when fully extended. Since the cables are subjected to only relatively small tensions, they can be nylon cord (or other nonconductor) to prevent inadvertent shorting of the antennas.

3.10.3 On some "yo-yo" despin devices utilized on previous Spacecraft, the weights have been tied to the internal structure and released by a guillotine mechanism. This often resulted in the problem of long cords flailing around inside the Spacecraft and tangling in the internal structure upon release. The problem was partially solved by using a retaining cord independent of the weights. This method, however, places the extra weight of the tie cord on only one of the despinning weights. (See figure 3-76.) A variation is to place a similar cord on the other weight and hold it in place by the first weight. Cubic proposed using a completely external tie independent of the weights. The tie would be nylon cord in high tension which, when severed, would snap off clear of the Spacecraft. This system has been tested successfully.

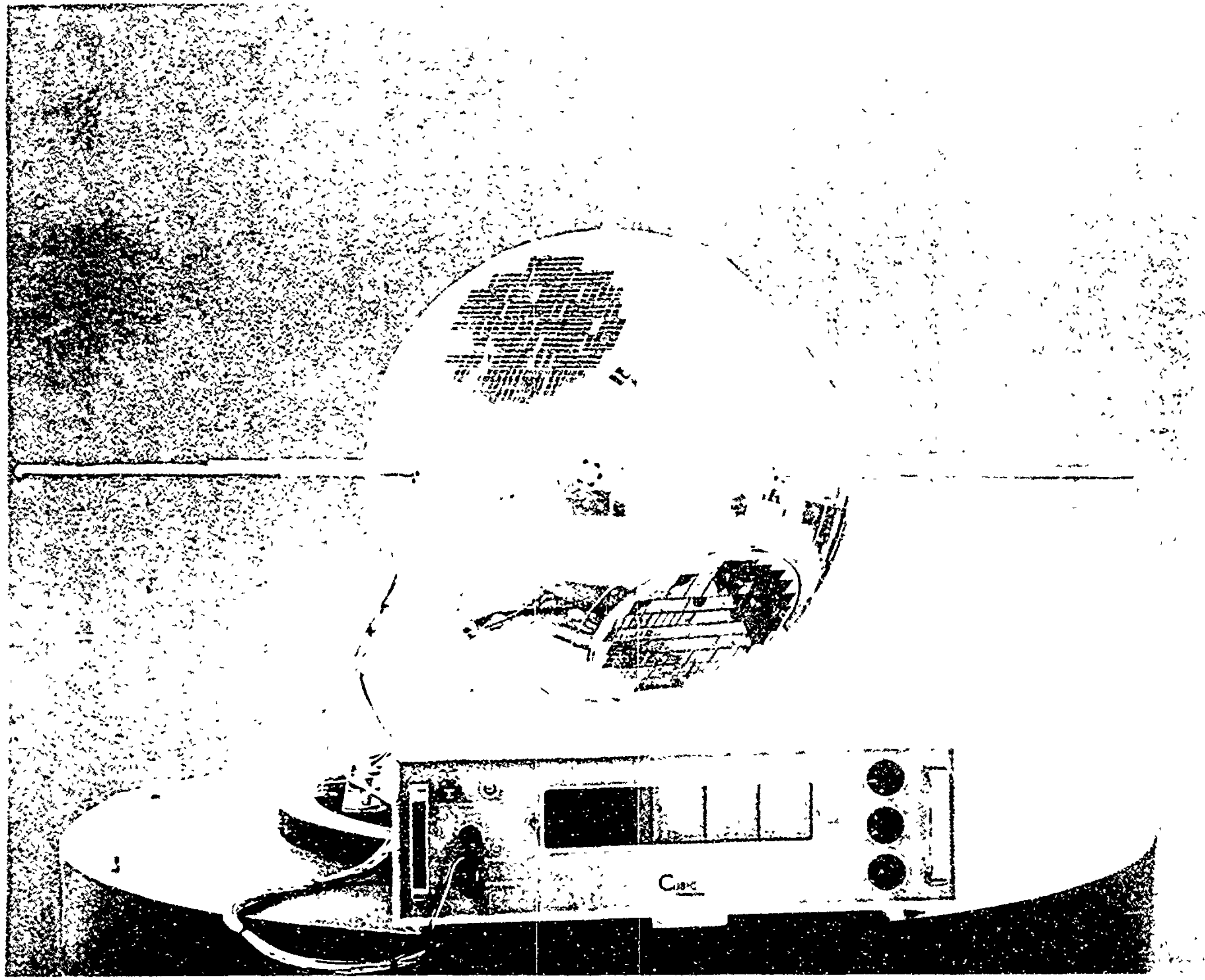
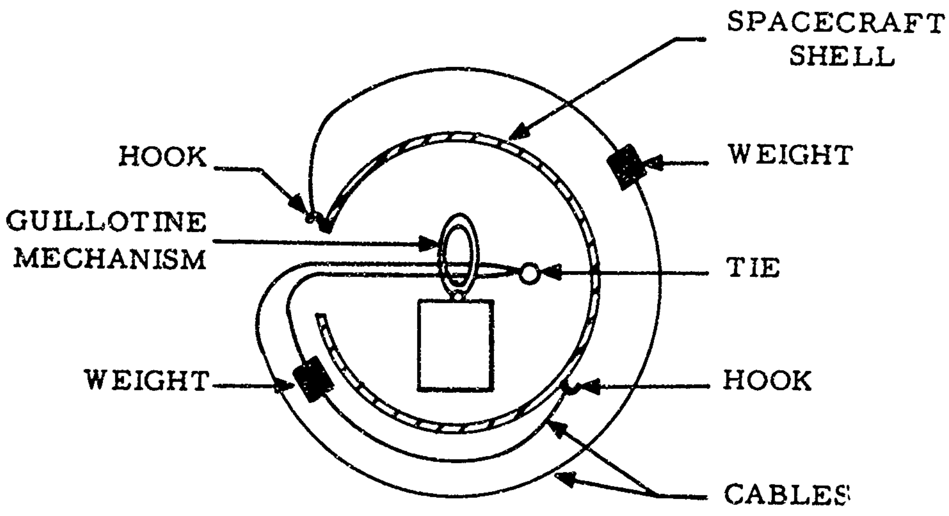
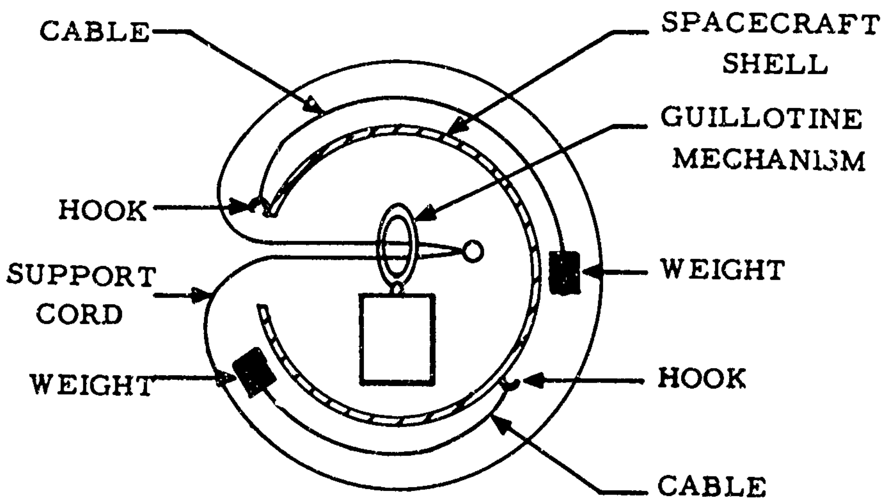


Figure 3-75. Use of Flight Plug for Monitoring Spacecraft Performance During Vibration Test



WEIGHTS TIED SEPARATELY



WEIGHTS TIED INDEPENDENTLY

Figure 3-76. Diagram of Optional Despinning Devices

3.11 Developmental History of Thermal Stabilization. Cubic conducted an investigation into the possible types of thermal coatings and performed a detailed mathematical analysis to determine the α/ϵ for the Spacecraft. The Physics Research Laboratory of the U. S. Army Engineer Research Laboratories confirmed Cubic's design and applied a coating of a silicon monoxide (SiO) to the satellite shell as a Government-furnished service.

3.11.1 General Problem. The sole purpose of this exterior surface finish is to establish the steady-state temperature of the Transponder and other electronics in the Spacecraft. Very briefly summarizing the underlying theory, any satellite receives energy from the sun, both by direct irradiation (when outside of the earth's shadow) and by relection from the earth's surface and atmosphere. For a 20-inch sphere, this irradiation is about 280 watts (direct) and up to 100 watts (by earth reflection), virtually all of which is in the band from 0.2 to 3.0 microns, with a peak at about 0.5 microns in the midvisible spectrum. The portion of this irradiation which is absorbed (that is, not reflected) is determined by absorbtivity α, an intrinsic property of the exterior surface. The satellite also loses heat by radiation to outer space in the manner of a "gray" body, at wavelengths of from about 4.0 to 50 microns, with a peak at 10 to 11 microns. The efficiency with which it accomplishes this is known, by comparison with a perfect radiator (or "black body") at the same temperature, as the emissivity ϵ. At temperature equilibrium, this radiation heat loss in the 4.0-to-50 micron range must balance the heat input in the 0.2-to-3.0 micron range; this clearly is governed by the ratio α/ϵ of the exterior surface. There are several practical methods for controlling the α/ϵ ratio to yield an acceptable interior temperature (0°C to 45°C) for the electronics.

3.11.1.1 This summary omits the effects arising from the presence of solar cells on about 20 per cent of the surface, variations in the orbit, the few watt-hours (per day) internally generated, and some long-wave radiation exchange between satellite and earth, all of which can be accounted for and fully controlled by a further refinement of the α/ϵ ratio. It is also well established that in a practical satellite, the thermal time constant between the outer shell and the interior electronics is long enough to suppress any temperature variations in the electronics during a nominal 100-minute orbit, part of which may be in the earth's shadow.

3.11.1.2 Temperature control is thus a practical problem of selecting a material (or a mixture of materials) having a desirable α/ϵ value. An intolerably hot satellite will result if $\alpha/\epsilon > 4$ (for example one made entirely of polished stainless steel). This condition will also be approached by almost any highly polished metal. Conversely, an $\alpha/\epsilon \ll 1$ will produce an intolerably cold satellite, something which would happen in a satellite completely covered by certain ceramics such as aluminum oxide. In between, values of α/ϵ from (approximately) 1.0 to 2.0 will produce acceptable temperatures.

3.11.2 Preliminary Considerations. One solution suggested is to provide a metallic satellite and coat portions of it with a ceramic in a pattern of stripes or polka-dots. This approach is that used for Explorer I, using a stainless steel shell covered 25 to 30 per cent with "Rokide A" aluminum oxide, an extremely tough and durable material. Telemetered temperatures adequately confirmed the performance of this mixed-materials design.

3.11.2.1 Another approach is exemplified by the Vanguard II satellite, in which a highly reflective aluminum surface (outermost) is coated with silicon monoxide having a carefully controlled thickness, which is quite transparent in the visible portion of the spectrum. Such a satellite becomes highly reflective to wavelengths of visual and photographic importance, as required by its mission. The result is also a very hot satellite except that SiO has a strong absorption (or emissivity) at 10 microns, the peak of the long-wavelength (cooling) radiation. Performance of this system has also proven highly satisfactory, as confirmed by telemetered interior temperatures. The technique of applying the proper thickness of SiO is advanced, employing a large vacuum evaporator and a complex variable-speed sphere rotating device within the chamber. Duplicate equipment probably does not exist outside Government agencies, and its provision would be expensive if the SiO-coating services were not government-furnished.

3.11.2.2 Note that since it plays no role in the final result, the gold plating and the over-coating of chromium and aluminum can be omitted, applying the SiO layer directly over a polished aluminum ball (as was done on Vanguard I). There is no

current justification for gold plating, except as a corrosion inhibitor from the time the shell is fabricated until it is coated with something else. In conclusion, since thermal design and the related SiO-coating can be supplied as GFE, an aluminum shell will be provided to facilitate the SiO-coating.

3.11.3 Development Details. The following paragraphs evaluate the factors affecting the thermal equilibrium of the Geodetic Spacecraft. Rigorous mathematical analysis of all factors of the problem are extensive and time-consuming. This part of the report involves a cursory mathematical treatment of the problem. The analysis established the conditions so that the maximum temperature does not exceed 308°K under the most severe conditions. It was extended to hold the minimum temperature above the 278°K for the minimum-sunlight orbits. The mathematical results were collated with the experimental results of other space vehicle programs to check the validity of the calculations. The following is a list of the pertinent assumptions and values used in the succeeding analysis.

(1) α/ϵ = 1.03 (surface)

(2) ϵ = 0.35 (assumed for calculation)

(3) α/ϵ = 0.95 (solar cells)

(4) Earth albedo = 50 per cent

(5) Satellite altitude = 50-1200 nautical miles

(6) Orbit = at any inclination

(7) Average power dissipation = 2 watts

(8) Predicted temperatures = $278^{\circ}K \quad T_p \quad ^{\circ}K \quad 308^{\circ}K$

(9) Shell and mount heat capacity = 5 kilocalories (assumed for calculations)

(10) Internal mass heat capacity at least 1.45 kilocalories (composed of nickel cadmium batteries weighing approximately 6.7 pounds, an aluminum transponder weighing 5.3 pounds, and an aluminum interior structure weighing not over 2 pounds).

(11) Metal conduction paths between shell and internal mass (consisting of direct contact at the south pole through the bottom thrust structure and separation fitting; direct contact at the north pole access port and direct contact through the main thrust structure to the antenna pads).

3.11.3.1 Surface Temperature. Aerodynamic heating may be neglected in temperature control problems associated with all vehicles in orbits above 300 km altitude (reference a). Then throughout the useful life of every vehicle of this series the heat balance will be maintained by radiative processes alone because no energy transfer can take place through conduction or convection to the surroundings.

3.11.3.1.1 Thermal design is a problem which will be solved here by simply choosing an exterior surface which has appropriate radiative properties. Design evaluations based on data from vehicles in orbit have justified the use of this simple procedure. Vanguard II results are of particular interest because its modification to the present spacecraft is such that extrapolation of that design (reference b) can be carried out with confidence. Hass, et al, (reference c) report that calculated and measured temperatures of the package did not differ by more than $11^{\circ}C$ and that temperature stabilization of the device appeared satisfactory. However, because of the great differences in the conduction paths between the shell and the interior of that vehicle and those of the Geodetic Spacecraft, these favorable results cannot be guaranteed by uncritical extrapolation. Power loss by any vehicle at the altitudes of interest is predicted by the Stefan-Boltzmann law:

$$\text{Power loss} = A\sigma\epsilon T^4 \qquad (3.2)$$

where:

A = outside surface area of the vehicle shell

σ = Stefan-Boltzmann constant = 5.67×10^{-12} watts $cm^{-2}\ K^{-4}$

ϵ = surface emissivity

T = vehicle surface temperature ($^{\circ}$K)

3.11.3.1.2 Application of Lambert's law leads to the well-known result that a reasonably uniform surface temperature over a sphere will cause it to absorb as a disc (for the same reason that the sun appears to be a disc). Thus the power received is proportional to (1) the cross-sectional area of the vehicle (the proportionality factor being called the "absorbtivity"), (2) the power per unit area at the vehicle from the sun's radiation (direct and reflected from the earth) and, (3) the earth's radiation. Solar radiation is from a source at an effective temperature of about 6,000°K while the earth's radiation is from a source at nearly the same temperature as the vehicle (about 255°K). Absorbtivities at the two widely different temperatures are not necessarily equal. Nonetheless, one simplification is possible in the case of the earth's radiation, where the approximate equality of the earth and vehicle temperatures permit use of Kirchhoff's law to set the absorbtivity equal to the emissivity. Summarizing the power input may be written as

$$\left\{ (P_{so} + P_{s1})a + P_e \epsilon \right\} \frac{A}{4} \qquad (3.3)$$

where:

P_{so} = power per unit area due to direct solar radiation incident at the vehicle = 0.135 watt/cm^2

P_{s1} = power per unit area due to earth-reflected solar radiation incident at the vehicle

P_e = power per unit area from earth radiation incident at the vehicle

a = absorbtivity of the vehicle surface for solar radiation.

The factor 1/4 comes about because the ratio of the cross-sectional to surface area is 4.

3.11.3.1.3 Thermal transients caused by dissipating the battery energy at an average rate in the order of milliwatts per square cm will at first be neglected compared to the power inputs and outputs which are of the order of watts.

NOTE

> Average power requirements will be less than two watts so that the flux through the surface required for equipment operation will be less than 1/4 milliwatt per square centimeter at equilibrium.

Consideration of the more significant transients due to entry into regions of shadow will also be postponed for later discussion. Measured temperatures aboard Vanguard II showed little short term variations and justify this procedure as surprisingly accurate. Then, following Hass, a mean temperature will be used and defined as the temperature that would obtain if steady state conditions prevailed. Under steady state conditions the power loss and gain are equal so that from equations (3.2) and (3.3) it follows at once that

$$(P_{so} + P_{sl})\frac{a}{\epsilon} + P_e = 4\sigma T_m^4 \qquad (3.4)$$

where:

T_m = mean temperature

Using the Stefan-Boltzmann law, Hibbs (reference b) has calculated the power received from the earth. His result is

$$P_e = 2\sigma T_E^4 (1 - \sqrt{2h}) \qquad (3.5)$$

where:

T_E = effective earth black body temperature = 255°K

h = vehicle altitude in units of earth radii.

Limiting altitude to the range 500 to 1200 nautical miles leads to

$$0.008 < P_e < 0.022 \text{ watts/cm}^2$$

3.11.3.1.4 Computation of the solar power received by reflection leads to integrals which can only be evaluated numerically (reference d). The wide range of orbits and orbital orientations which must be considered for the Geodetic Spacecraft makes this type of analysis prohibitively expensive because for thermal design purposes, highly accurate reflective predictions are not warranted. Instead an approximation given by Hibbs (reference b) was used for the solar power received by reflection which may be written

$$P_{s1} = 2\rho_E P_{so} (1 - \sqrt{2h}) \cos \theta_s \qquad (3.6)$$

where:

ρ_E = coefficient of diffuse reflection from the earth

θ_s = angle between an earth diameter extended through the vehicle and one extended through the sun.

Inspection of equation (3.6) predicts no reflected power reception for an orbit in a plane normal to the direction of the sun. Although the earth-reflected power does not completely vanish in regions outside the earth's shadow, this lower limit will nonetheless be used. The maximum will be taken as the average given by (3.6) in the sunlit hemisphere when the orbital plane passes through the sun so that

$$P_{s1} < \frac{4}{\pi} \rho_E P_{so} (1 - \sqrt{2h}) < 0.040 \text{ watts/cm}^2 \qquad (3.7)$$

where the numerical value on the right was obtained by assuming an albedo of 50 per cent and a minimum altitude of 500 nautical miles.

3.11.3.1.5 Solving equation (3.4) for α/ϵ is valid if the vehicle is in the sunlight long enough to reach equilibrium. The result is

$$\left(\frac{\alpha}{\epsilon}\right)_o = \frac{4\sigma T_{max}^4 - P_e}{P_{so} + P_{sl}} \tag{3.8}$$

where the subscript o is used to emphasize the fact that an upper limit of the temperature is only reached if the vehicle attains thermal equilibrium. From the preceding results, equation (3.8) leads to the conclusion that for T_{max} equal to 293^oK, the ratio for a vehicle in full sunlight lies in the range

$$0.83 < \alpha/\epsilon < 1.18.$$

If transit of the satellite through the shadow takes the same time as that required for equilibrium to be attained a minimum temperature will be reached which will be determined by the conditions for a balance between the radiation received from the earth and that emitted to all space.

3.11.3.1.6 Equation (3.4) with solar power equated to zero together with equation (3.5) show this minimum is given by

$$T_{min} = \left\{\frac{1}{2}\left(1 - \sqrt{2h}\right)\right\}^{\frac{1}{4}} T_E \tag{3.9}$$

and brings out the fact that the lower temperature extreme is independent of the surface (a consequence of Kirchhoff's law). Unfortunately this results in rather low equilibrium temperatures which, for orbits considered here, are in the range

$$135 < T_{min} < 176^oK.$$

3.11.3.1.7 Unacceptably low shadow equilibrium temperatures make mandatory a more detailed investigation of the behavior of the vehicles when their orbits include regions of shadow. If the temperature gradient through the shell may

be neglected, the difference between the heat input and output which is no longer zero becomes proportional to the time rate of change of the temperature. The proportionality factor is the total "heat capacity." Using equations (3.2) and (3.3), there results a situation described by

$$\left\{(P_{so} + P_{sl}) a + P_e \epsilon \right\} A/4 - A\sigma\epsilon T^4 = C \frac{\partial T}{\partial t} \quad (3.10)$$

where:

C = total heat capacity

t = time

At the instant of entry into the earth's shadow the solar heat input vanishes and equation (3.10) becomes

$$\left.\frac{\partial T}{\partial t}\right)_o = \frac{\epsilon A}{4C}\left[P_e - 4\sigma T_o^4\right] \quad (3.11)$$

where the subscript o indicates conditions at entry of the shadow. Taking as representative an emissivity of 0.35, an initial temperature of 293°K, and a total heat capacity of 5 kilocalories per Kelvin degree results in

$$\left.\frac{\partial T}{\partial t}\right)_o < 0.005^\circ K/sec \quad (3.12)$$

Note in equation (3.9) that T must decrease so that this estimate is an upper limit of the rate of change of temperature. Then it can be guaranteed that the temperature will drop less than 15°C during passage through the shadow in any orbit orientation.

3.11.3.1.8 Vehicle surface temperature rate of change will be given by

$$\left.\frac{\partial T}{\partial t}\right)_1 = \frac{A\epsilon}{4C}\left\{P_{so} + P_{sl}) a/\epsilon + P_e - 4\sigma T_1^4\right\} \quad (3.13)$$

at the instant of reentry into the region of sunlight. Equation (3.13) was obtained by adding the power from the sun to equation (3.2) and the subscript 1 was affixed to denote conditions at the start of a heating cycle.

3.11.3.1.9 If a/ϵ should be taken as its lowest value (0.83) to avoid overheating while the conditions for minimum heating actually obtain, equation (3.13) shows that the temperature would not exceed 273°K in 100 per cent sunlight. A 15°K drop in temperature in the shadow would then lead to an initial heating rate of 0.008°K/sec which would be sufficient to raise the shell back to its equilibrium temperature in about half an hour. This estimate is based on the fact that although the heating rate is not accurately a linear function of time an inspection of equation (3.13) shows it is a reasonable assumption until the equilibrium temperature is approached rather closely.

3.11.3.1.10 From the preceding discussion it appears That a better design approach would be to allow the shell to approach the upper limit in the 100 per cent sunlit, maximum heating situation. Returning to equation (3.8), the ratio a/ϵ then turns out to be almost exactly unity. The interior maximum temperature will of course be somewhat less. This matter is taken up in detail later. In the shadow the cooling rate starts out at about 0.006°K/sec by equation (3.11). For a maximum of about one hour in the dark, a pessimistic estimate of the cooling is about 20°K. In the case of minimum heating the equilibrium upper temperature would be 282°K with an associated maximum cooling rate of 0.0035°K per second. The lower temperature limit would then be above 270°K for a pass through the shadow taking one hour with the cooling being less than 12°K.

3.11.3.2 <u>Interior Temperature</u>. Hibbs (reference b) has shown that for Explorer configurations the rate of radiative heat transfer between the shell and the interior mass is negligible compared to that of conductive transfer. His result may be written

$$(T_{in})_{rad} = \frac{\sigma}{C_{in}} \left(T_{out}^{4} - T_{in}^{4}\right) \qquad (3.14)$$

where:

$(T_{in})_{rad}$ = time rate of change of the interior temperature due to equilibration by radiaton.

C_{in} = total heat capacity of the interior mass

and the subscripts "out" and "in" denote conditions at the shell surface and the interior mass, respectively. In the present problem the interior mass is about 14 pounds. The average heat capacity will certainly be greater than that of aluminum which leads to

$$C_{in} > 6.3 \text{ kilowatt-sec}/^{\circ}\text{K}.$$

3.11.3.2.1 Taking the most extreme range estimated for the temperature variation results in

$$(\dot{T}_{in})_{rad} < 0.004\ ^{\circ}\text{K/hour}.$$

Since there is no convection this result indicates "the equation of heat conduction",

$$\nabla^2 T = \frac{1}{a^2} \quad \frac{\partial T}{\partial t} \tag{3.15}$$

where:

$$a^2 = \text{thermal diffusivity}.$$

This very accurately decribes the temperature distribution in space and time throughout the interior.

3.11.3.2.2 Certain geometries, such as the one analyzed by Hibbs, lead to a realistic one-dimensional "slab solution". When two regions at relatively definite temperatures are separated by thin ("thin" compared to cross-sectional area) insulating washers this well-known solution may be used with confidence. On the other hand, where all metallic conduction paths exist between the shell and the interior, as in the Geodetic Spacecraft, even rough estimates based on the "slab solution" would be highly suspect.

3.11.3.2.3 The geometry is such that gradients exist not only along the supporting structural members but also over the shell surface extending out from all points of interior-exterior contact. This is much more complicated than the situation where all points of contact are through insulators. In that

case it is perfectly legitimate to assume gradient temperature vanishes everywhere outside the space occupied by the insulator. It is not believed that equation (3.15) has been solved for even simple geometries that approach those of the Geodetic Spacecraft. Such a solution, based on an acceptable model, would call for a major effort and was not attempted. Although equation (3.15) appears completely unmanageable for this application, there are boundary conditions for which the problem is tractable. One such boundary value problem involves the transfer of heat to the interior of a semi-infinite homogenous slab which is heated at the surface in such a way that the surface temperature variation is given by

$$\Delta T(O) = \Delta T_o \cos \omega t \qquad (3.16)$$

where:

$\Delta T(\theta)$ = temperature change at a given distance below the surface

ΔT_o = amplitude of the surface temperature variation

x = distance below the surface.

3.11.3.2.4 At first sight these boundary conditions would not appear to be of particular interest. Nonetheless, they have been applied with success as approximations to the geophysical problem of determining the variation of temperature with depth due to solar heating. In some ways that problem is similar to that of the heating and cooling of the Spacecraft. The worst case for temperature excursions in the spacecraft interior occurs when the vehicle spends the maximum time (approaching half a period) in shadow. Here the assumption of sinusoidal variation of surface temperature can be defended about as well as the same assumption for the heating of the surface of the earth. The ratio of conduction path cross-section areas to shell surface area is very small so the assumption of a semi-infinite conduction path does not at first seem justified. However, since the conduction paths are very long (in terms of conduction path cross-section dimensions) temperature variations across the path will be negligible compared to those along the path. Penetration is thus well approximated by a one-dimensional

solution (i.e., is equivalent to assumption of a semi-infinite medium). The approximation is quite valid in the long linkages from the shell to the main internal mass of the Spacecraft. However, where the conduction path is through the main mass the problem again becomes three-dimensional and the one-dimensional assumption breaks down. Nonetheless, since the large heat capacity of the interior mass will dampen the amplitude of the temperature excursions, the semi-infinite assumption in this region may be considered conservative for our purposes in spite of its lack of accuracy.

3.11.3.2.5 Equation (3.15) applied to the geophysical problem involving sinusoidal surface heating as given by equation (3.16) has a well-known solution (reference e) which may be written for the present application as

$$\Delta T(x, t) = \Delta T_o \exp - \left\{ \frac{x}{a} \left(\frac{\omega}{2}\right)^{\frac{1}{2}} \right\} X$$

$$\mathrm{Cos} \left\{ \omega t - \frac{x}{a} \left(\frac{\omega}{2}\right)^{\frac{1}{2}} \right\} \qquad (3.17)$$

From equation (3.17) it is evident that the dispersion which arises through the frequency dependence of the velocity of propagation of the temperature disturbance is

$$a\,(2\omega)^{\frac{1}{2}}.$$

For magnesium the thermal diffusivity is

$$a^2 = 1.0 \text{ cm}^2/\text{sec}$$

giving the temperature wave a velocity of propagation into the interior somewhat less than a millimeter per second for an orbit with maximum time in the shadow (assuming a period of 2 hours).

3.11.3.2.6 Of greater importance in this problem is the fact that equation (3.17) predicts an exponential decay of the temperature variation with depth. Before applying the results, a selection of the amplitude ΔT_o must be made. No effort was made to base the selection on a rigorous analysis. Rather, a qualitative

discussion of the surface temperature variations is given to establish its value adequately for design purposes.

3.11.3.2.7 Gradients over the surface are such that the region in the neighborhood of conduction path termination at the surface will be cooler during the hottest time of the heating half-cycle because of conduction to the interior. During the cooling half-cycle when minimum shell temperatures occur the gradients are such that heat is conducted over the surface away from the conduction terminations requiring that the conduction path terminal temperatures be higher than the shell average. Assumption of a shell temperature excursion amplitude equal to that obtaining for a perfectly conducting shell (i.e., no surface gradients) will thus be conservative for purposes of estimating the temperature excursion of the interior.

3.11.3.2.8 Taking the worst case, where the vehicle approaches half a period in darkness in a two-hour orbit with the surface coated for conditions of maximum heating, the temperature variation will be estimated in the event that the lower limits of radiant power reception actually obtain. Then the temperature as predicted by (3.17) at the center of the sphere the temperature would be well within the range

$$275 < T_{center} < 282^{\circ}K$$

It will be noted that at the minimum, the estimate indicates failure to meet specification by cooling 3° too low but keeping in mind the extreme pessimism of the conditions assumed and the conservatism of calculation it appears certain that the design will meet specifications for minimum heating situations. In fact if the temperature of the solar cells does not fall significantly below $273^{\circ}K$, the design will take care of losses due to the area covered by the solar cells. (In the event that the solar cells emissivity is much above 0.35, allowance will have to be made for the increased heat loss during the cooling cycle.)

3.11.3.3 Conclusions. Following the analysis, Cubic recommended an SiO coating which would provide an α/ϵ ratio of 1.03. This coating would keep the temperature of the Spacecraft within the $20^{\circ} \pm 15^{\circ}C$ range. The spacecraft temperature approaches a maximum of $35^{\circ}C$ under maximum-sunlight conditions. It is not expected to reach $+5^{\circ}C$ even under minimum-sunlight conditions.

3.11.4 Fabrication. Prior to applying the SiO coating to the prototype Spacecraft, the U. S. Army Physics Research Laboratory checked Cubic's analysis and recommendations. Dr. Hass informed Cubic that their assumption of 0.35 for the emissivity of aluminum was overconservative. This meant that Cubic's estimate of the cooling in shadow was overly pessimistic. Dr. Hass stated that an ϵ = 0.18 could be provided for aluminum; he was in agreement with Cubic's procedure for specifying the surface based on the 100 per cent sunlit orbits. Dr. Hass requested a recalculation of the α/ϵ for the aluminum structure, considering the effect of the solar cells and white paint on the outer surface of the Spacecraft. As the value for the ϵ of aluminum was given to be = 0.18, Cubic calculated the value for α as follows:

$$\left(\frac{\alpha}{\epsilon}\right)_{GSC} = \frac{(A_{sp})(\alpha_{sp}) + (A_{sc})(\alpha_{sc}) + (A_p)(\alpha_p)}{(A_{sp})(\epsilon_{sp}) + (A_{sc})(\epsilon_{sc}) + (A_p)(\epsilon_p)} \qquad (3.18)$$

where:

$\left(\frac{\alpha}{\epsilon}\right)_{GSC}$ = 1.03 ($\frac{\alpha}{\epsilon}$ for the entire Geodetic Spacecraft)

A_{sp} = 896 in^2 (Area of aluminum surface)

A_{sc} = 297 in^2 (Area of solar cells)

A_p = 57 in^2 (Area of paint)

$\frac{\alpha_{sp}}{\epsilon_{sp}} = \frac{(0.18)}{\epsilon}$ (α/ϵ for aluminum surface)

$\frac{\alpha_{sc}}{\epsilon_{sc}} = \frac{0.78}{0.83}$ (α/ϵ for solar cells)

$\frac{\alpha_p}{\epsilon_p} = \frac{0.33}{0.89}$ (α/ϵ for paint)

The resultant α/ϵ for aluminum was determined to be 1.45, with an $\alpha = 0.18$ and $\epsilon = 0.1238$.

3.11.4.1 The U. S. Army Physics Research Laboratory discovered that the 0.63 finish on the exterior of the prototype Spacecraft was not satisfactory for the application of the SiO. The laboratory polished the shell and Cubic informed Brooks and Perkins of the correct degree of finish for the second and third Spacecraft. The U. S. Army Research Laboratory coatings were applied to each shell in the following sequence: 1200 Angstrom units of SiO, 700 Angstrom units of aluminum, and 6000 Angstrom units of SiO.

3.11.5 α/ϵ Testing. A test of the α/ϵ for the Spacecraft was performed at the Applied Physics Laboratory of Johns Hopkins University in Silver Springs, Maryland. As the α/ϵ test is considered to be a part of the environmental testing, the details are covered under this heading.

3.12 Developmental History of Static and Dynamic Balancing. The Electronic Balancing Co., of Long Beach, California performed the static and dynamic balancing of the three Geodetic Spacecraft.

3.12.1 General Problem. The general problem was to accomplish a static balance of each Spacecraft within the allowable center of gravity displacement of less than 0.005 inches, and to determine its dynamic performance at 160 to 30 rpm.

3.12.2 Preliminary Considerations. Cubic stipulated that they would perform all handling of the Spacecraft and would attach the weights as specified by Electronic Balancing Co. Electronic Balancing Co. provided a specially prepared room with two horizontal balancers and one vertical balancer. Figure 3-77 shows a typical balancing setup. Prior to the static balance, Cubic measured the center of gravity of the Spacecraft. The cg was determined to be 0.5 inch below the equator.

3.12.3 Development Details. Spacecraft No. 1 required a bottom balance of 233 gr and a top balance of 275 gr. Wraps of solder placed around the vertical trusses were used for the initial static balance.

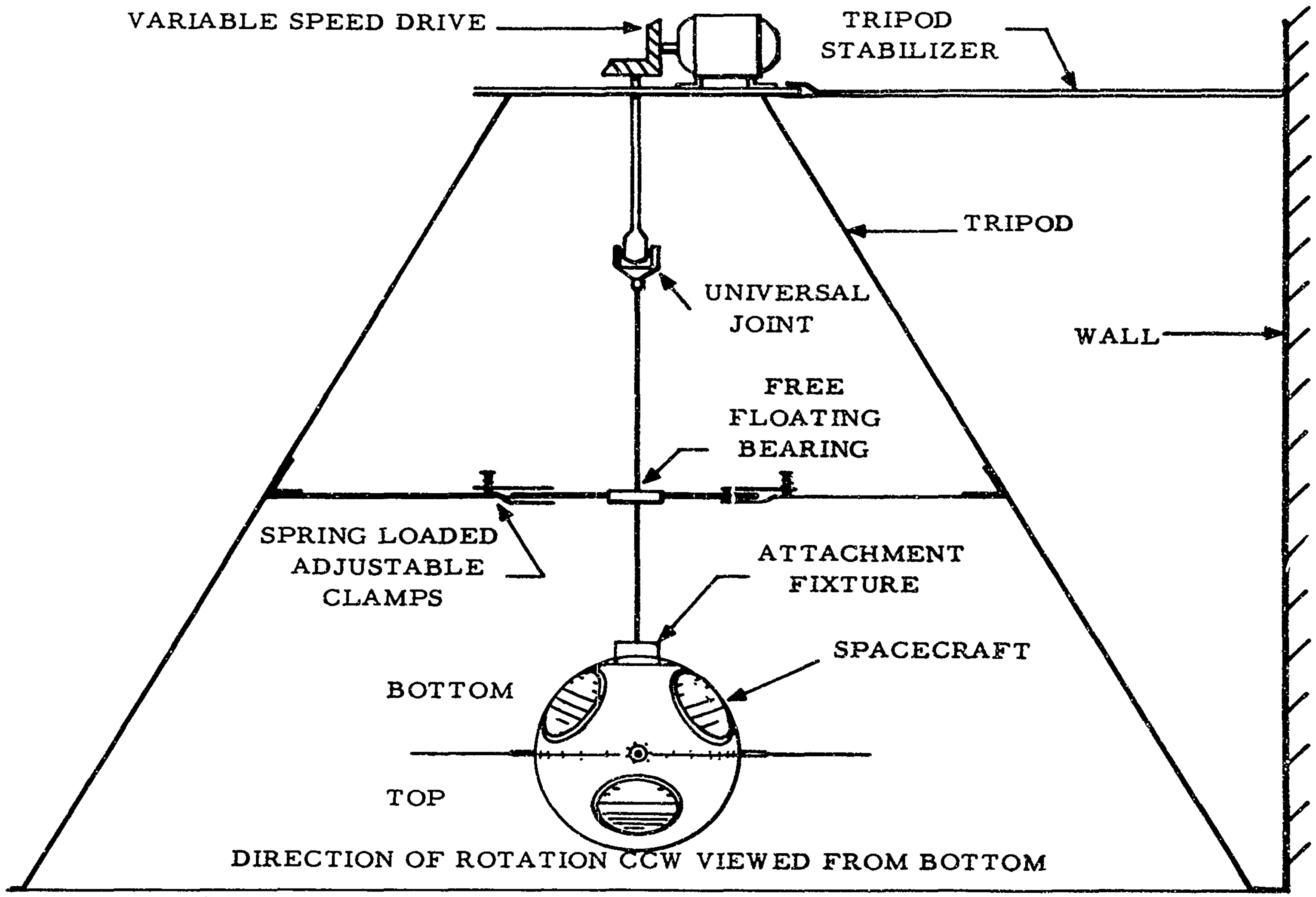


Figure 3-77. Typical Setup for Dynamic Balancing of Spacecraft

These were replaced with split bronze rings which are clamped to the vertical trusses on the side nearest the flight plug. The balance at both the top and the bottom of the Spacecraft is at a runout of approximately 1/16-inch at 250-300 rpm. The north pole antenna and the top access plate runs out of true by approximately 1/4 inch. If the top access plate hole pattern is used as an indication of unbalance, the 1/4-inch discrepancy appears to be due to the fact that the north pole antenna is not precisely at the geometric axis of the Spacecraft.

3.12.3.1 During the balancing of Spacecraft No. 1, it was decided that in subsequent balancing operations, the internal structure, not including the antennas, would be balanced first. Then the shell, antennas, and solar cell plaques would be added and the final balancing performed.

3.12.3.2 During the balancing of Spacecraft No. 3, when the frame and the shell were assembled and rotated, balance could not be achieved. The solar cell plaques were removed from the shell, the assembly was rotated, and balance was achieved. Upon weighing the solar cell plaques, three were found to weigh exactly 269 gr; the remaining three weighed 308gr, 304gr, and 287gr, respectively. The three equivalent plaques were installed on the north hemisphere. Weights were added to the remaining plaques to bring them all up to 308gr. The weighted plaques were installed on the south hemisphere, the assembly was rotated, and balance was achieved.

3.12.3.3 Prior to final balance of the second Spacecraft, the solar cell plaques were removed, weighed, and balanced.

3.13 Developmental History of Environmental Testing. After conducting a study of environmental information gathered from NASA, NRL, and APL, Cubic planned a test program to include vibration testing (both random and sinusoidal), acceleration testing, α/ϵ testing, and thermal-vacuum testing. Ryan Electronics, San Diego, California performed the vibration tests; General Dynamics, San Diego Division conducted all acceleration tests and the thermal-vacuum tests on Spacecraft Nos. 1 and 2; and the Applied Physics Laboratory of Johns Hopkins University, Silver Springs, Maryland performed an α/ϵ test and the thermal-vacuum test on Spacecraft No. 3.

3.13.1 General Problem. Cubic's proposed environmental test program separated the three Geodetic Spacecraft into a prototype unit, a flight unit, and a backup flight unit. The most severe testing was to be performed on the prototype unit in order to qualify the design. Less severe testing would be conducted on the flight and backup flight models to ensure successful performance in orbit without the danger of fatiguing the equipment. However, at GIMRADA's direction, the program was revised to provide identical environmental tests for all three Spacecraft.

3.13.1.1 The revised detailed test plan was designed to insure that each of the Spacecraft will successfully and reliably operate in space when launched by either a Scout or an Able-Star rocket. Considering the worst possible environmental conditions which could be encountered, the experience provided by other programs, and factors affecting the reliability of spacecraft and the ultimate success of the project, Cubic proposed the following tests:

(1) Vibration - Test all units in the three major axis to levels experienced in the Scout vehicle (includes both random and sinusoidal)

(2) Acceleration - Test all units to 35g's in direction of launch thrust with 1g in the transverse axis.

(3) Thermal-Vacuum - Accomplish thermal-vacuum testing to prove thermal design and capability of the unit to operate in a space environment.

(4) α/ϵ - A test to verify the absorptivity/emissivity ratio of the thermally coated Spacecraft. This test program is believed best to insure success of the project, as it is similar to programs used by other government agencies primarily engaged in spacecraft development.

3.13.2 Preliminary Considerations. The following paragraphs list the details of the proposed environmental test program for the Geodetic Spacecraft.

3.13.2.1 Testing of Components and Packages. The following tests shall be conducted on all components and packages, the design of which has not been previously tested and proved.

(1) Vibration tests - In accordance with table XIII. Units under test will be mounted on the test fixture with the major horizontal axis oriented in the same relative position as when mounted in the Spacecraft. Axes definitions are shown in figure 3-78.

(2) Low temperature thermal tests - Exposure to $0^{\circ}C$ at one atmosphere for a period of one hour.

(3) High temperature thermal tests - Exposure to $+50^{\circ}C$ at one atmosphere for a period of one hour.

Components and packages shall operate satisfactorily throughout and subsequent to the tests in accordance with criteria established by the Project Engineer.

3.13.2.2 Testing of Assembled Geodetic Spacecraft Payload. Cubic designed and fabricated a baseplate and retaining ring to simulate the vehicle portion of the separation mechanism. (See figure 3-79.) During all the environmental tests the baseplate was secured to the existing facility test fixtures, and the Spacecraft was clamped to the plate by the retaining ring. Cubic utilized the identical arrangement for purposes of securing the Spacecraft within the shipping containers. The tests described in the following paragraphs were planned for each of the completely assembled Spacecraft.

3.13.2.2.1 Vibration Tests. These tests shall be conducted in accordance with table XIV. The axes definitions are given in figure 3-78.

3.13.2.2.2 Acceleration Tests. The assembled payload shall be subjected for a period of one minute to an acceleration of 35 g in the direction of launch and a force of 1 g in one transverse direction, applied concurrently.

TABLE XIII

VIBRATION ENVIRONMENT, COMPONENTS AND PACKAGES

Sinusoidal Vibration					
Longitudinal					
Frequency Range (cps)	5-50	50-500	500-2000	2000-3000	3000-5000
Sweep Time (min)	1.66	1.66	1.00	0.50	0.50
Peak Acceleration (g)	2.3	10.7	21.0	54.0	21.0
Transverse, Two Axis					
Frequency Range (cps)	5-50	50-500	500-2000	2000-3000	3000-5000
Sweep Time (min)	1.66	1.66	1.00	0.60	0.50
Peak Acceleration (g)	0.9	2.1	4.2	17.0	17.0
Random Noise Vibration					
Longitudinal 4 Minutes					
Frequency Range (cps)	5-200	200-400	400-2000		
Gaussian Random Noise (g^2/cps)	0.12	-12 db/ octave	0.01		

TABLE XIII (Cont)

Transverse, 2 Axes, 4 Minutes Each			
Frequency Range (cps)	5-25	25-100	100-200
Gaussian Random Noise, (g^2/cps)	0.60	-12 db/ octave	0.01

3.13.2.2.3 Space/Thermal Test. The Geodetic Spacecraft shall be irradiated under the following prescribed conditions until the internal temperatures reach equilibrium, and for an additional two hours at equilibrium conditions.

(1) Conditions - Radiation intensity of 450 ±50 Btu/hr/ft^2 uniformly distributed over the plane of the Spacecraft within ±10 per cent of the average intensity; spectral distribution shall correspond to solar energy within ±10 per cent from 0.30 to 2.5 microns.

(2) Chamber - Evacuated to 1×10^{-5} mm Hg or less with walls cooled to -202°F (-130°C) or colder. Chamber capability wall temperature, if warmer, may be accepted by the Project Engineer.

(3) Spacecraft Mounting - Spacecraft shall be mounted with spin axis vertical and rotated 1.0 ±0.5 rpm about its spin axis.

The external and internal skin temperatures, as well as the temperature at the instrument support tube, must be recorded throughout the run. Limits of these temperatures will be in accordance with criteria established by the Project Engineer.

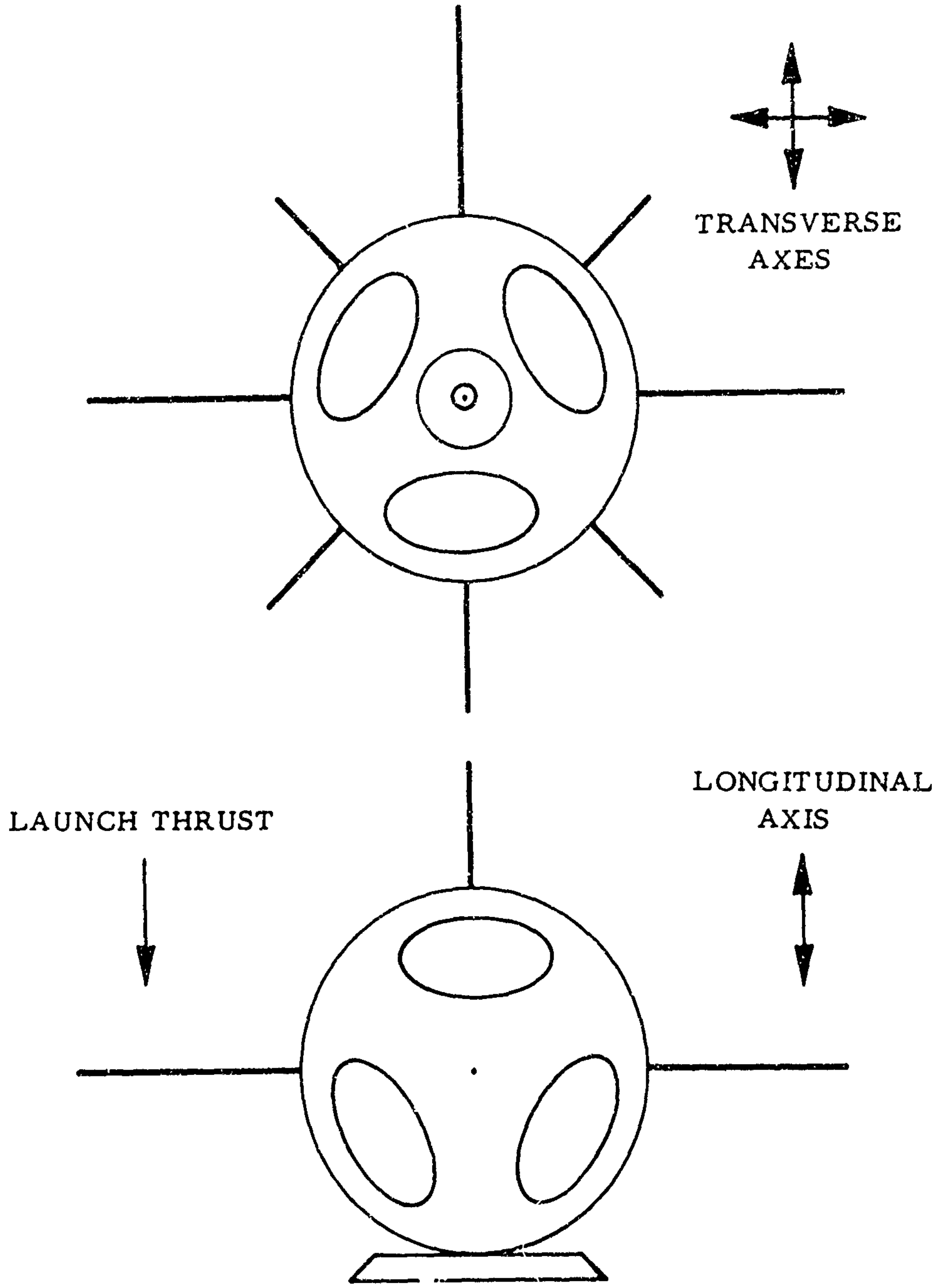


Figure 3-78. Axis Definitions for Environmental Tests

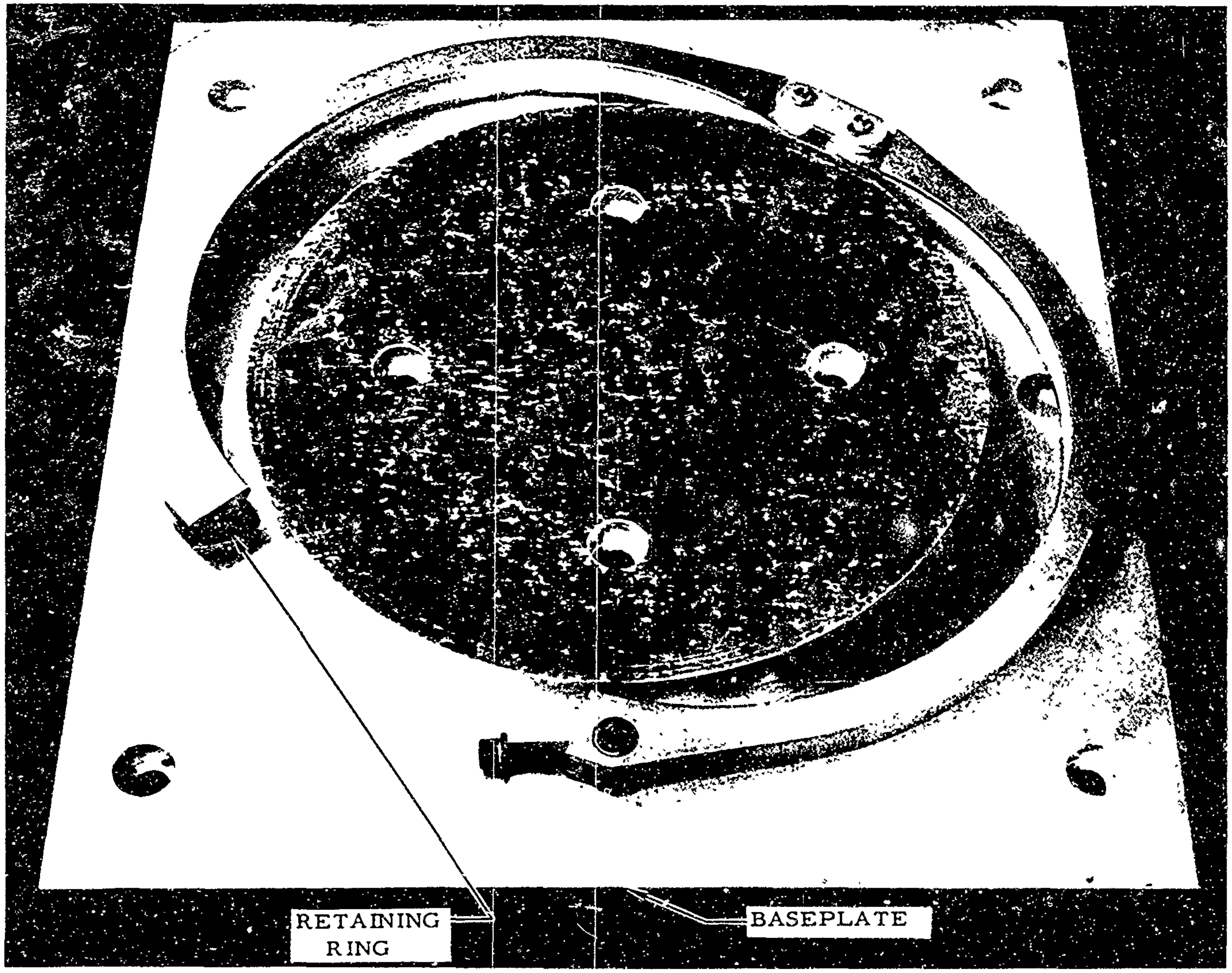


Figure 3-79. Baseplate and Retaining Ring

TABLE XIV

VIBRATION ENVIRONMENT, ASSEMBLED PAYLOAD

Sinusoidal Vibration					
Longitudinal					
Frequency Range (cps)	50-50	50-500	500-2000	2000-3000	3000-5000
Sweep Time (min)	1.66	1.66	1.00	0.50	0.50
Peak Acceleration (g)	1.5	7.3	14.0	36.0	14.0
Transverse, 2 Axes					
Frequency Range (cps)	5-50	50-500	500-2000	2000-3000	3000-5000
Sweep Time (min)	1.66	1.66	1.00	0.60	0.60
Peak Acceleration (g)	0.6	1.4	2.8	11.3	11.3

Random Noise Vibration			
Longitudinal 4 Minutes			
Frequency Range (cps)	5-200	200-400	400-2000
Gaussian Random Noise (g^2/cps)	0.08	-12 db/ octave	.0066

TABLE XIV (Cont)

Transverse, 2 Axes, 4 Minutes Each			
Frequency Range (cps)	5-25	25-100	100-2000
Gaussian Random Noise (g^2/cps)	.40	-12 db/ octave	.0066

3.13.2.2.4 Internal/External Thermal Test. The Geodetic Spacecraft shall be installed in a chamber evacuated to 1×10^{-5} mm Hg.

3.13.2.2.4.1 The internal temperature of the Spacecraft shall be stabilized at $\pm 5^{\circ}$C. The Spacecraft shell temperature shall then be cycled through the maximum excursion attainable in 100 minutes peak-to-peak while maintaining the internal temperature within the limits of $+15^{\circ}$C and -5°C. Four cycles about the mean temperature shall be made.

3.13.2.2.4.2 The internal temperature of the Spacecraft shall be stabilized at $+35^{\circ}$C. The Spacecraft shell temperature shall then be cycled through the maximum excursion attainable in 100 minutes peak-to-peak while maintaining the internal temperature within the limits of $+45^{\circ}$C and $+25^{\circ}$C. Four cycles about the mean temperature shall be made. Limits of shell temperature shall be in accordance with criteria established by the Project Engineer.

3.13.2.2.4.3 The electronic equipment contained in the Spacecraft shall operate satisfactorily throughout and subsequent to these tests in accordance with criteria established by the Project Engineer.

3.13.2.3 α/ϵ Test. A test to verify the absorptivity/emissivity ratio of one Spacecraft shall be conducted. The vacuum chamber shall be evacuated to a pressure of 1×10^{-5} mm of Hg or better and the chamber walls cooled to approximately -90°F. A black body ($\alpha/\epsilon = 1$) shall be rotated within the chamber to provide the average wall temperature as seen by a sphere. Arc lamps simulating the sun shall illuminate the body until temperature equilibrium is reached to provide the reference data. The Spacecraft shall be similarly rotated and exposed in the chamber until temperature equilibrium occurs in order to provide data for determination of the ratio.

3.13.3 Development Details - Testing Components. The printed circuit boards designed and fabricated for the telemetry subsystem were subjected to the environmental testing outlined in paragraph 3.13.2.1. Following the initial vibration tests at Ryan Electronics, San Diego, California on 3 May 1961, the boards were rebuilt to provide larger contact eyelets. In subsequent vibration tests, each of the redesigned boards met the test criteria, and the components operated satisfactorily following the tests. The following paragraphs describe the vibration tests in detail as recorded by Ryan Electronics:

(1) Test items mounted in one of two transverse axes, and subjected to the following sinusoidal vibration: (See figure 3-80.)

5-50 cps at ±0.9 g for 1 minute, 30 seconds

50-500 cps at ±2.1 g for 1 minute, 25 seconds

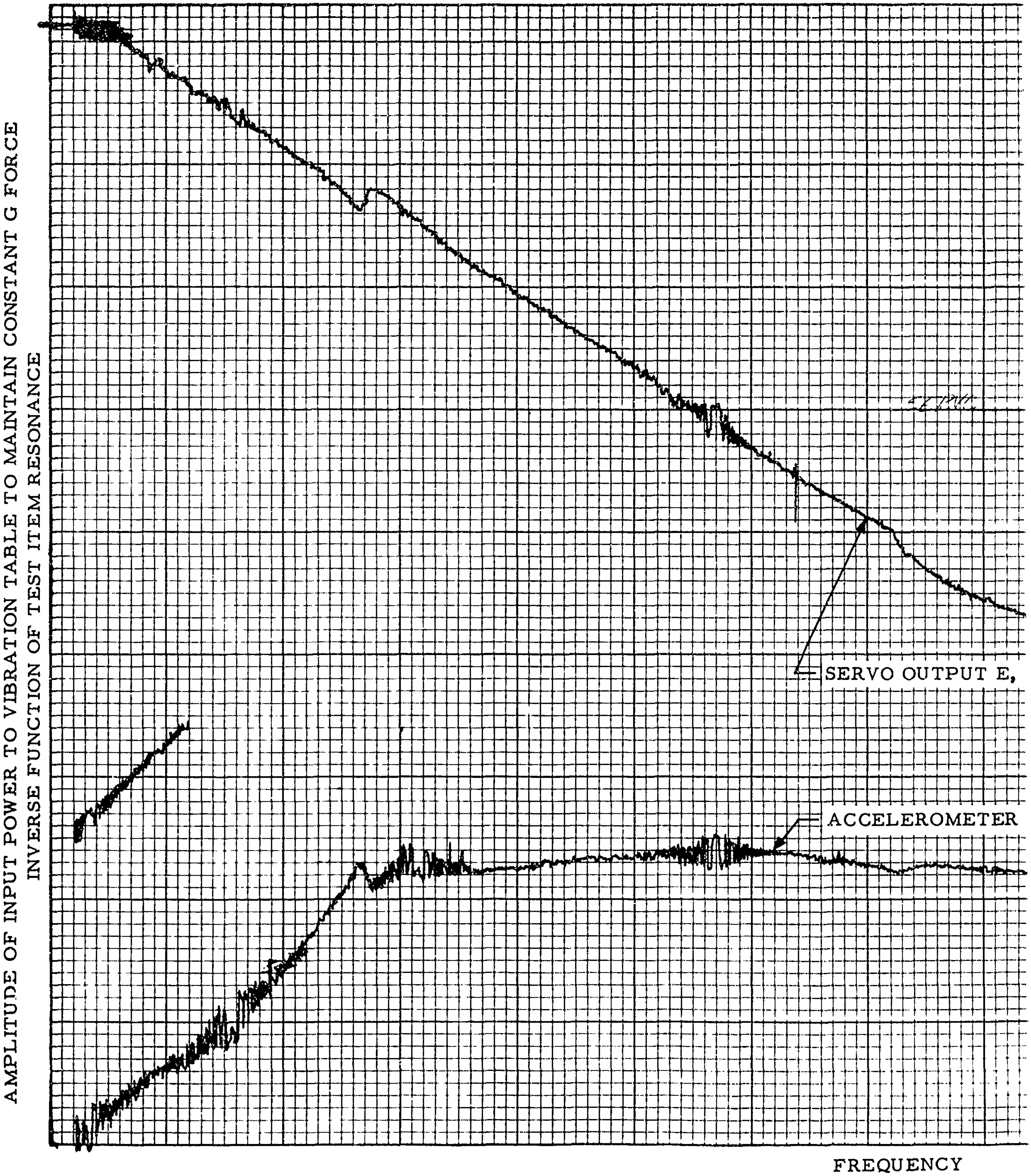
500-2000 cps at ±4.2 g for 1 minute, 5 seconds

2000-3000 cps at ±17 g for 35 seconds

50-2000 cps at ±10 g for 1 minute, 55 seconds

(2) Test items and test mount ~~then~~ equalized to within ±1.5 db and subject~~ed~~ to the following random vibration for a period of four minutes.

15-25 cps at 3.8 g (rms), 1/2 inch D. A. and 11.4 g (rms) clipping


AMPLITUDE OF INPUT POWER TO VIBRATION TABLE TO MAINTAIN CONSTANT G FORCE
INVERSE FUNCTION OF TEST ITEM RESONANCE
SERVO OUTPUT E,
ACCELEROMETER
FREQUENCY

A

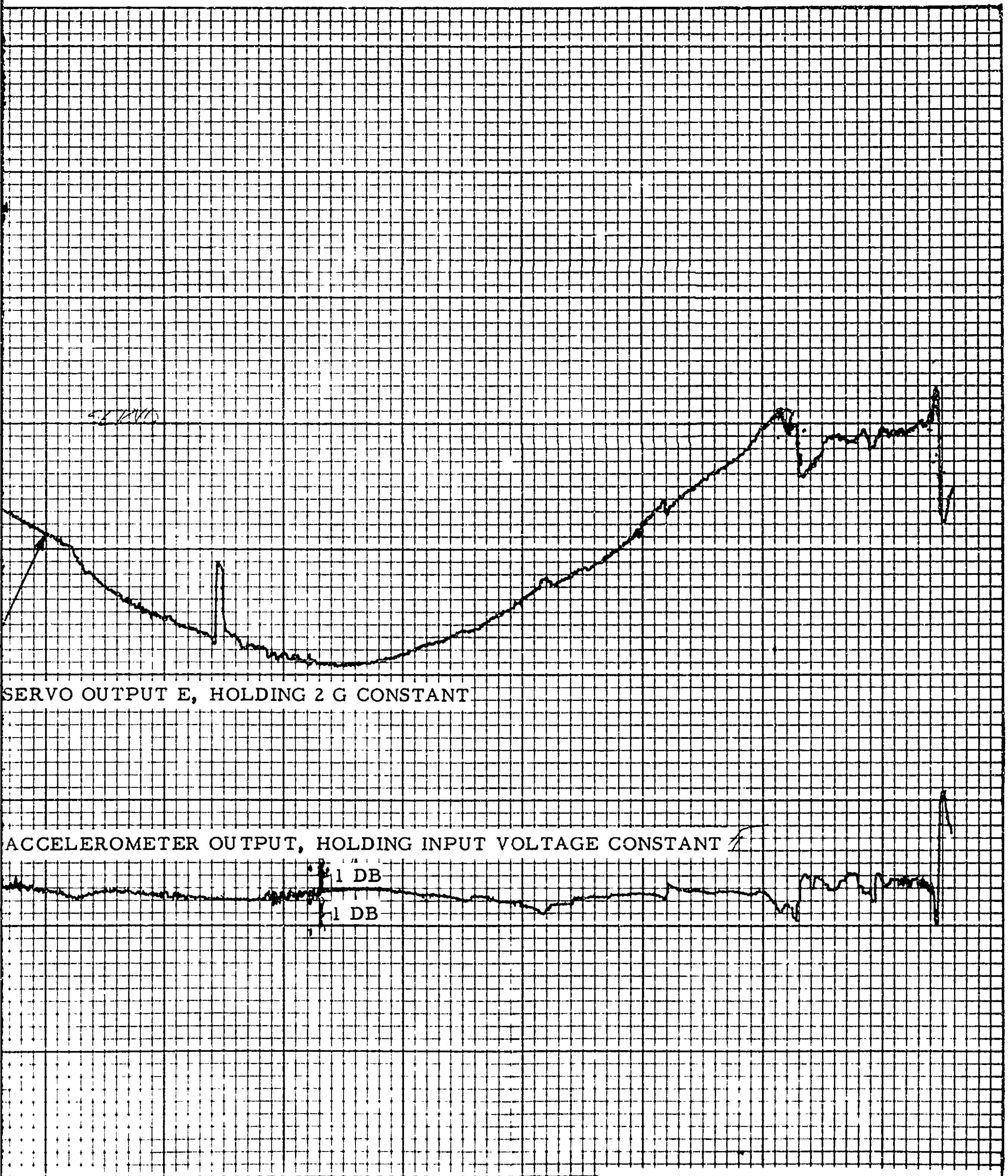


Figure 3-80. Telemetry Printed Circuit Board Resonant Response, First Transverse Axis

25-100 cps - 15 db/octave ~~for 4 minutes~~

100-2000 cps at 4.35 g (rms), 1/2 inch D. A. and 13.05 g (rms) clipping

(3) Test items mounted in second transverse axis and subjected to the following sinusoidal vibration: (See figure 3-81.)

5-50 cps at ±0.9 g for 1 minute, 30 seconds

50-500 cps at ±2.1 g for 1 minute, 35 seconds

500-2000 cps at ±4.2 g for 1 minute, 9 seconds

2000-3000 cps at ±17 g for 35 seconds

50-2000 cps at ±10 g for 1 minute, 55 seconds

(4) Test items and mount then equalized to within ±1.5 db and subjected to the following random vibration:

15-25 cps at 3.8 g (rms), 1/2 inch D. A. and 11.4 g (rms) clipping

25-100 cps -15 db/octave for 4 minutes

100-2000 cps at 4.35 g (rms), 1/2 inch D. A. and 13.05 g (rms) clipping

(5) Test items mounted in longitudinal axis, and items and mount equalized to within ±1 db

(6) Test items subjected to the following random vibration: (See figure 3-82.)

15-200 cps at 4.8 g (rms), 1/2 inch D. A. and 14.4 g (rms) clipping

200-400 cps -19 db/octave for 4 minutes

400-2000 cps at 4.0 g (rms), 1/2 inch D. A. and 1.2 g (rms) clipping for 4 minutes.

(7) Test items then subjected to the following sinusoidal vibrations:

5-50 cps at ±0.9 inch D. A. or ±2.3 g for 1 minute, 30 seconds

50-500 cps at ±0.7 g for 1 minute, 50 seconds

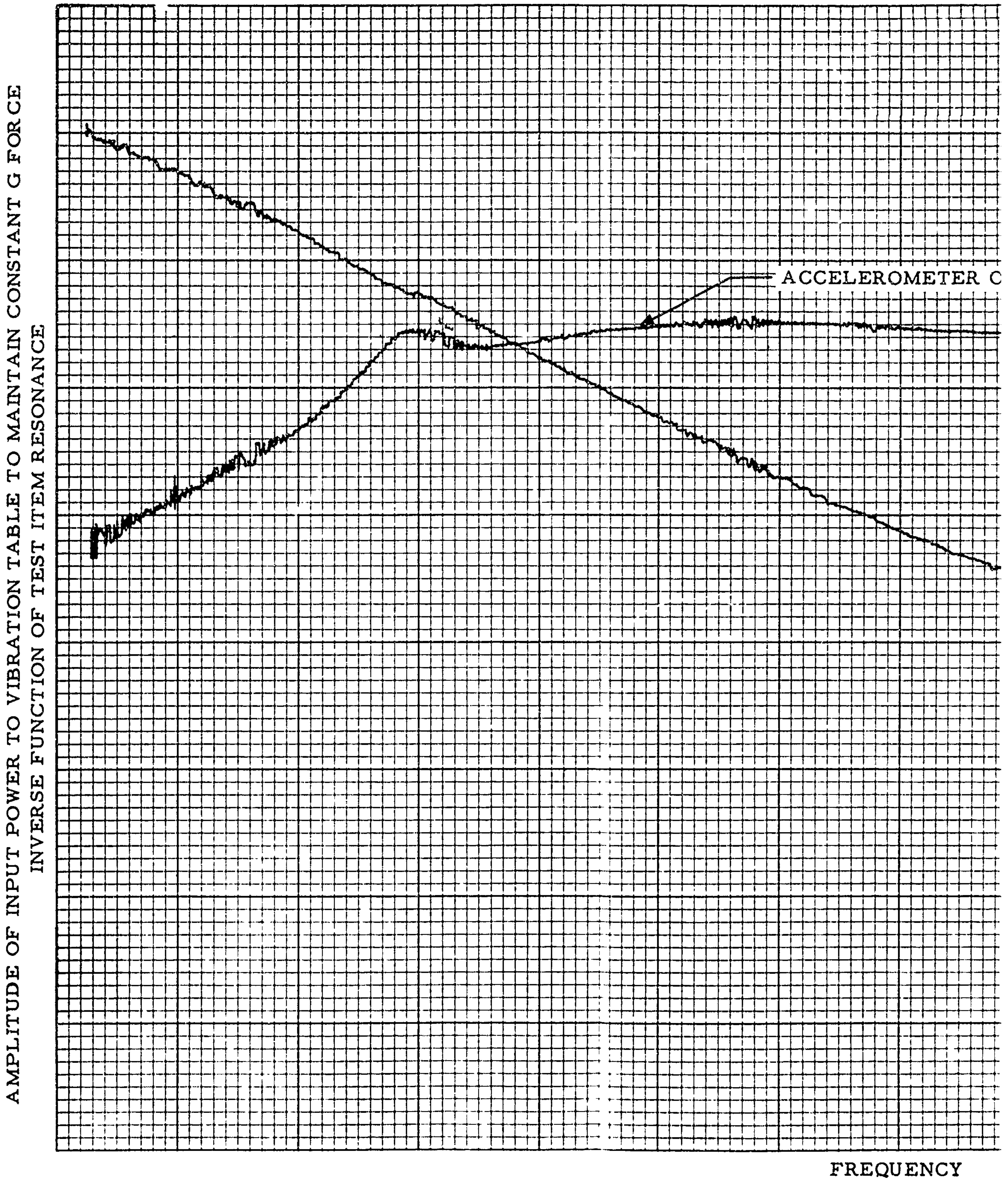
500-2000 cps at 2.1 g for 1 minute, 2 seconds

2000-3000 cps at ±21 and ±30 g for 35 seconds

3.13.4 Development Details - Testing Assembled Spacecraft. The proposed environmental tests were conducted on all three Spacecraft at various test facilities. Refer to table XV for a schedule of the complete program. As the tests varied somewhat, the following paragraphs discuss in sequence the individual tests performed on each Spacecraft. In general, available reponse curves are included in the body of the discussion. The test data from which the curves were derived appears as an appendix.

3.13.4.1 Description of Tests for Spacecraft No. 1. Geodetic Spacecraft No. 1 has successfully completed the environmental test program. This consisted of random and sinusoidal vibration tests, acceleration tests and thermal-vacuum tests.

3.13.4.1.1 Random and Sinusoidal Vibration Tests. The vibration tests discussed in the following paragraphs were performed at Ryan Electronics, San Diego, California from 19 May 1961 to 22 May 1961. The Spacecraft was tested to the Scout environment in the three major axes. Both random and sinusoidal vibration tests were performed over the ranges of 5 to 3,000 cps, limited only by the capability of the shakers. Refer to table XVI for a list of the test equipment utilized. See figure 3-74 for the test setup. The spacecraft was nonoperational during this test. The procedure was as follows:


AMPLITUDE OF INPUT POWER TO VIBRATION TABLE TO MAINTAIN CONSTANT G FORCE
INVERSE FUNCTION OF TEST ITEM RESONANCE
ACCELEROMETER C
FREQUENCY

A

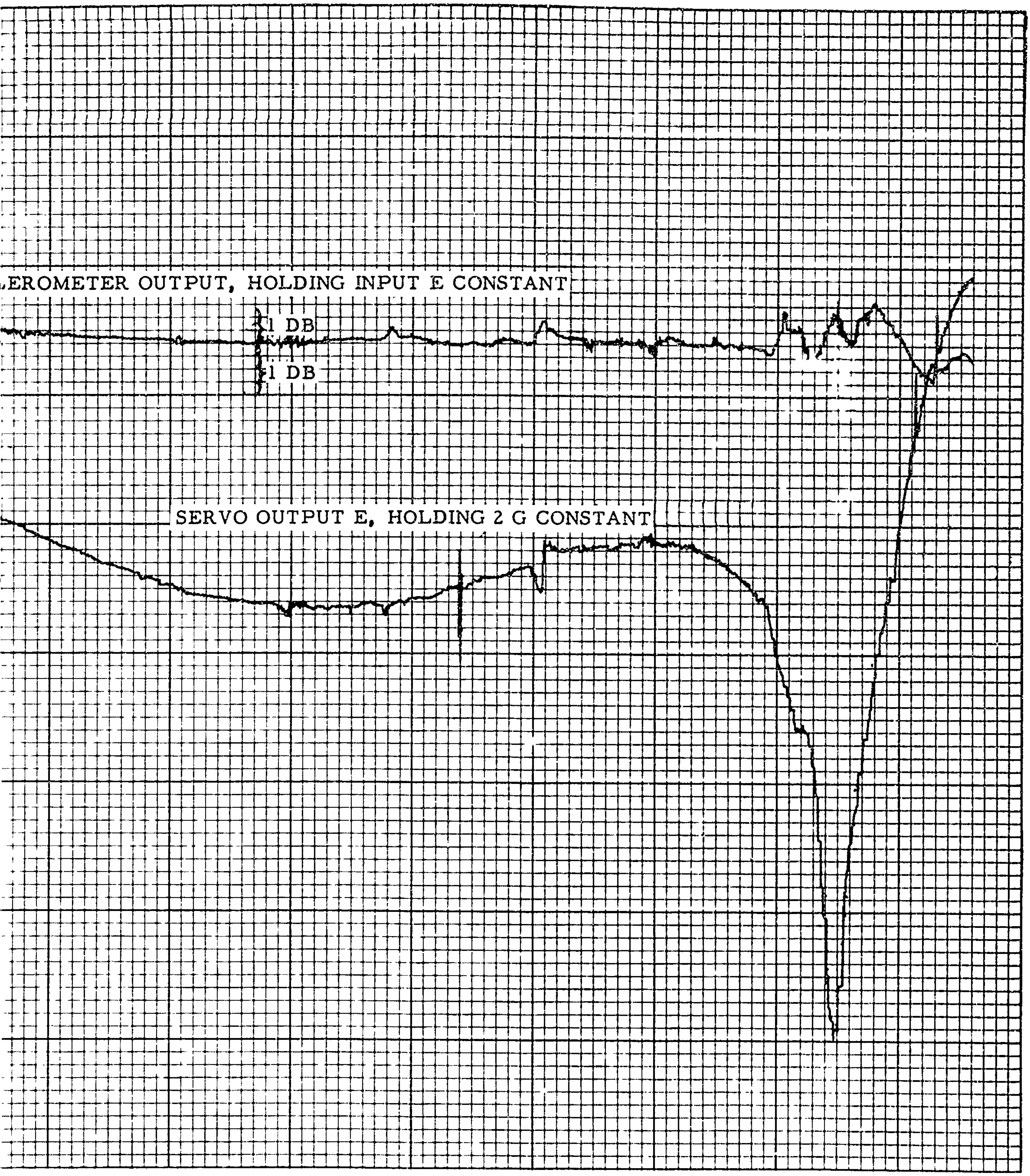
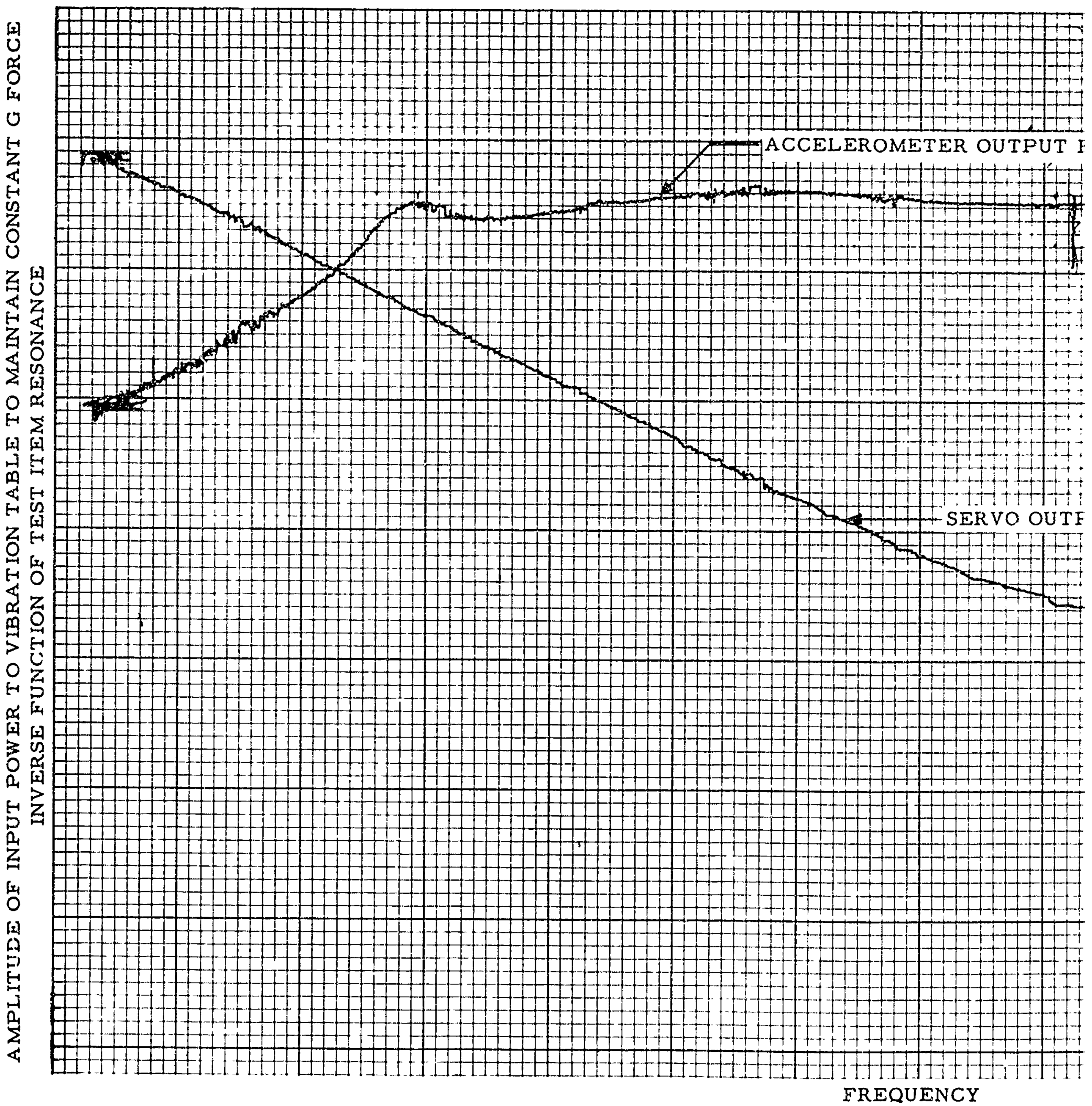


Figure 3-81. Telemetry Printed Circuit Board Resonant Response, Second Transverse Axis


AMPLITUDE OF INPUT POWER TO VIBRATION TABLE TO MAINTAIN CONSTANT G FORCE
INVERSE FUNCTION OF TEST ITEM RESONANCE
ACCELEROMETER OUTPUT
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FREQUENCY

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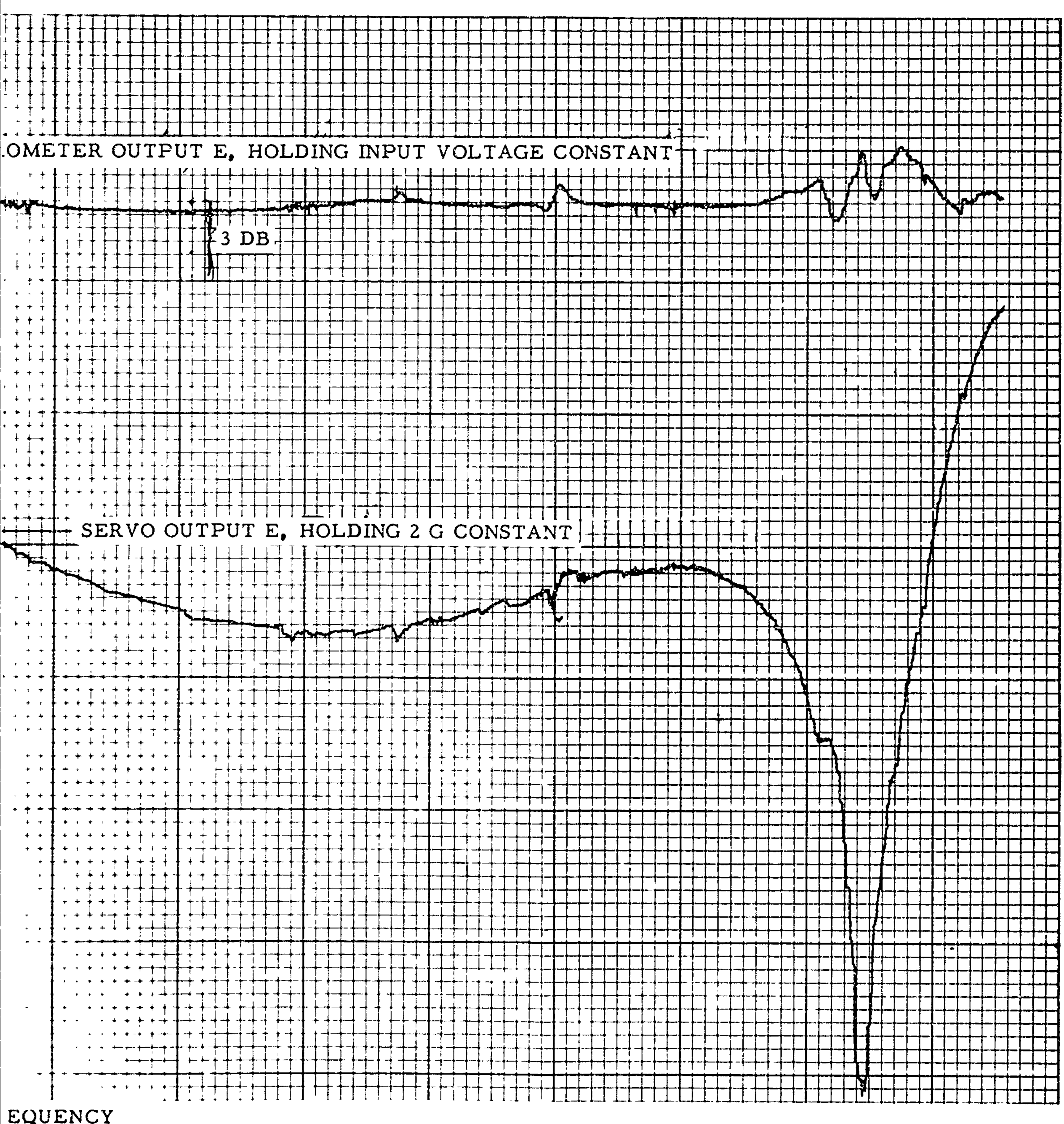


Figure 3-82. Telemetry Printed Circuit Board Resonant Response, Longitudinal Axis

TABLE XV

SCHEDULE OF ENVIRONMENTAL TESTING
FOR
ASSEMBLED GEODETIC SPACECRAFT

Date	Test and Facility	Spacecraft No.
22 May	Vibration testing at Ryan Electronics	1
25 May	Acceleration testing at General Dynamics, San Diego Division	1
27 May through 31 May	Thermal-Vacuum testing at General Dynamics, San Diego Division	1
20 July	Vibration testing at Ryan Electronics	3
21 July	Acceleration testing at General Dynamics, San Diego Division	3
4 August	Vibration testing at Ryan Electronics	2
7 August	Acceleration testing at General Dynamics, San Diego Division	2
7 August through 11 August	Thermal-vacuum testing at General Dynamics, San Diego Division	2
18 August	α/ϵ and thermal-vacuum testing at Applied Physics Laboratory	3

TABLE XVI

TEST EQUIPMENT UTILIZED FOR VIBRATION TESTS

Description	Manufacturer	Model No.
Vibration Exciter	Ling Electronics	177A
Electronic Power Supply	Ling Electronics	PP30/40
Control Console	Ling Electronics	R1007
Accelerometer	Endevco Corporation	2213
Vibration Slip Table	Ling Electronics	WM-450

(1) The test specimen was secured to a test fixture and mounted on the vibration exciter in a position to apply vibration along the longitudinal axis. The frequency of vibration was then scanned from 5 to 50 cps at 0.95 inch D. A. or ±3g (whichever limited in 1.66 minutes); 50 to 500 cps at ±7.3g in 1.66 minutes; 500 to 2000 cps at ±14g in 1 minute; and from 2000 to 3000 cps at ±36g or to the limits of the vibration machine in 0.5 minute. The frequency of vibration was swept at an approximate logarithmic rate.

(2) With the test specimen still mounted in the longitudinal axis, the vibration exciter response was equalized to within ±2 db of a flat power spectral density curve within the frequency band of 100 to 2000 cps. The acceleration clipping was adjusted so that 99.7 per cent of the true Gaussian distribution was realized and displacement clipping was adjusted to 0.5 inch D. A. The test specimen was then subjected to random vibration with a frequency band of 100 to 200 cps at a 4g (rms) level, down -16 db/octave from 200 to 400 cps, for 2 minutes. The bandwidth was readjusted and the test specimen was then subjected to random vibration with a frequency band of 400 to 2000 cps at a 3.3g (rms) level for 2 minutes.

(3) The test specimen (and fixture) was then mounted to the vibration exciter in a position to apply vibration along the first transverse axis. The frequency of vibration was then scanned from 5 to 50 cps at ±0.6g in 1.66 minutes; 50 to 500 cps at ±1.4g in 1.66 minutes; 500 to 2000 cps at ±2.8g in 1 minute; and from 2000 to 3000 cps at ±11.3 g or to the limits of the vibration machine in 0.60 minute. The frequency of vibration was swept at an approximate logarithmic rate.

(4) With the test specimen still mounted in the first transverse axis, the vibration exciter response was equalized to within ±3 db of a flat power spectral density curve within the frequency band of 100 to 2000 cps. The acceleration clipping was adjusted so that 99.7 per cent of the true Gaussian distribution was realized and displacement clipping was adjusted to 0.5 inch D. A. The test specimen was then subjected to random vibration with a frequency band of 100 to 2000 cps at a 3.6g (rms) level for 4 minutes.

(5) Paragraphs (3) and (4) above were then repeated in the second transverse axis.

The vibration test was completed as described in the test procedure. During and at the conclusion of the vibration test, the test specimen was examined for mechanical damage. As no evidence of mechanical damage was apparent and the electronics performed satisfactorily after the test, it was concluded that the Spacecraft had successfully withstood the specified vibration levels.

Since a resonant condition was noted in the frequency range of 30-32 cps, it was decided that the separation fixtures would be modified on serial numbers 2 and 3. Refer to paragraph 3.4.3.1 for modification details.

.1.2 Acceleration Tests.
neral Dynamics, San Diego
ιy 1961. The Spacecraft
ı the thrust axis with 1 g
›eriod. Acceleration was
.he one-minute period to bring
the centrifuge up to speed. Figures 3-83 and 3-84 show details of the test-mounting in the centrifuge.

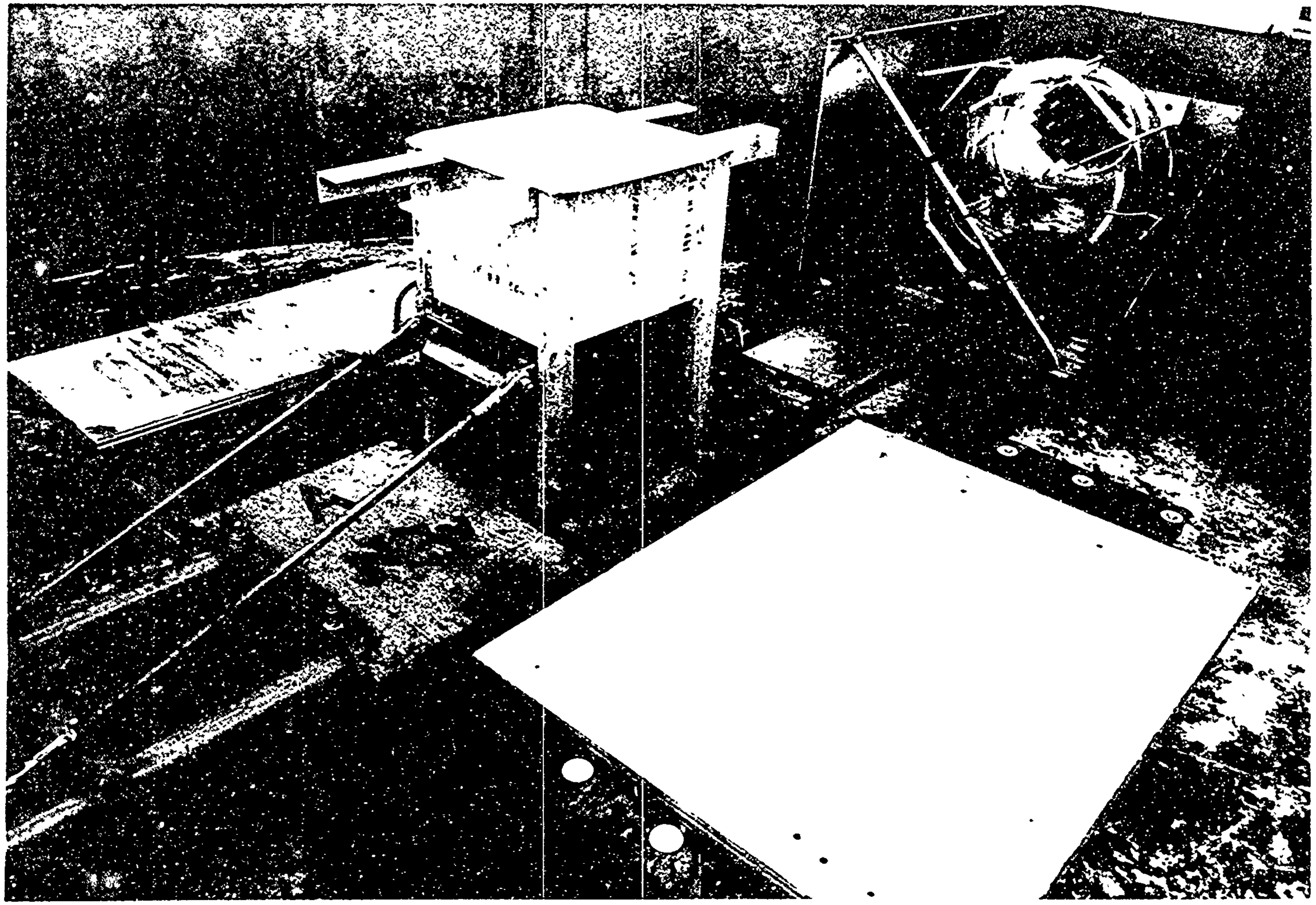


Figure 3-83. Geodetic Spacecraft Mounted in Centrifuge

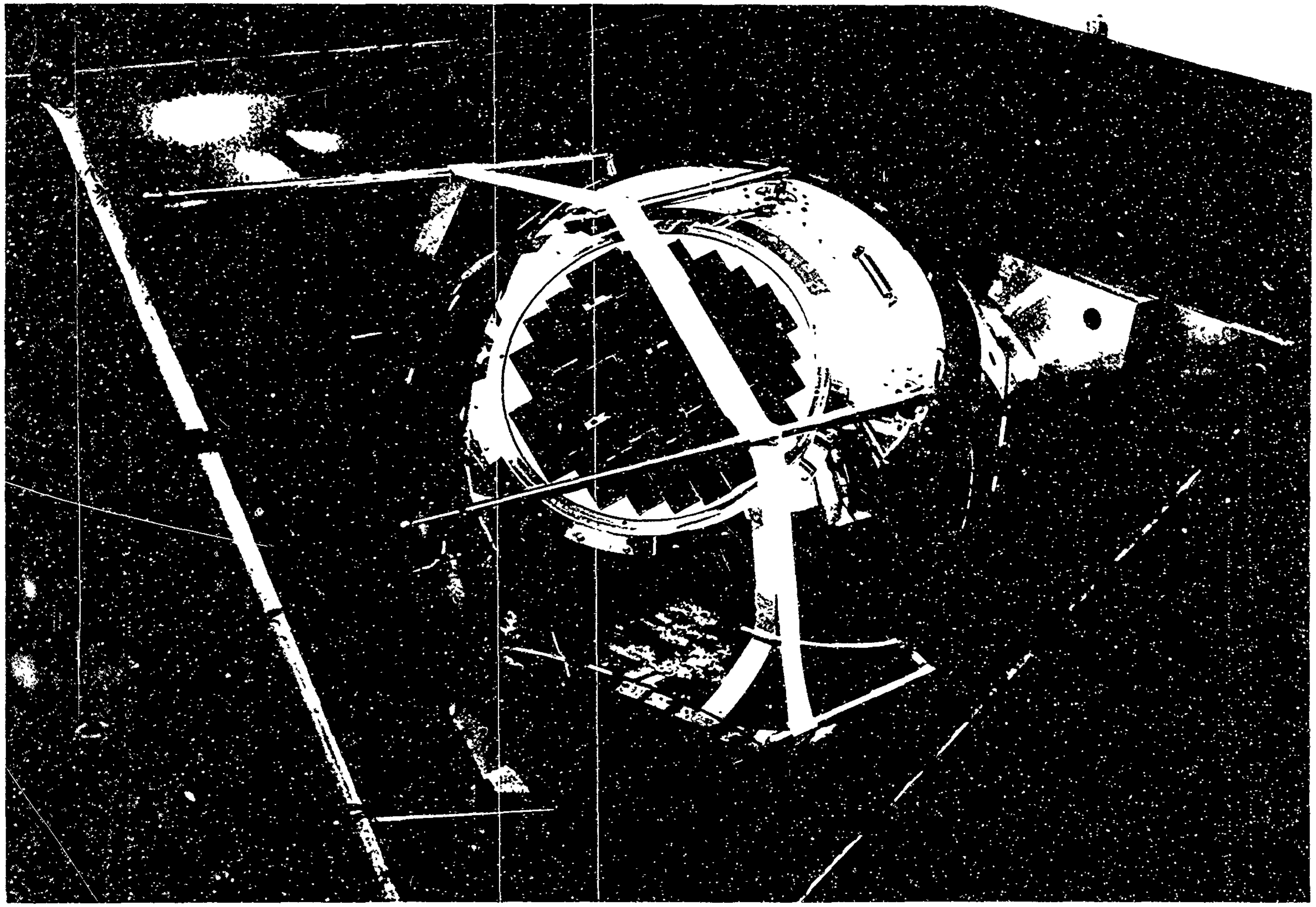


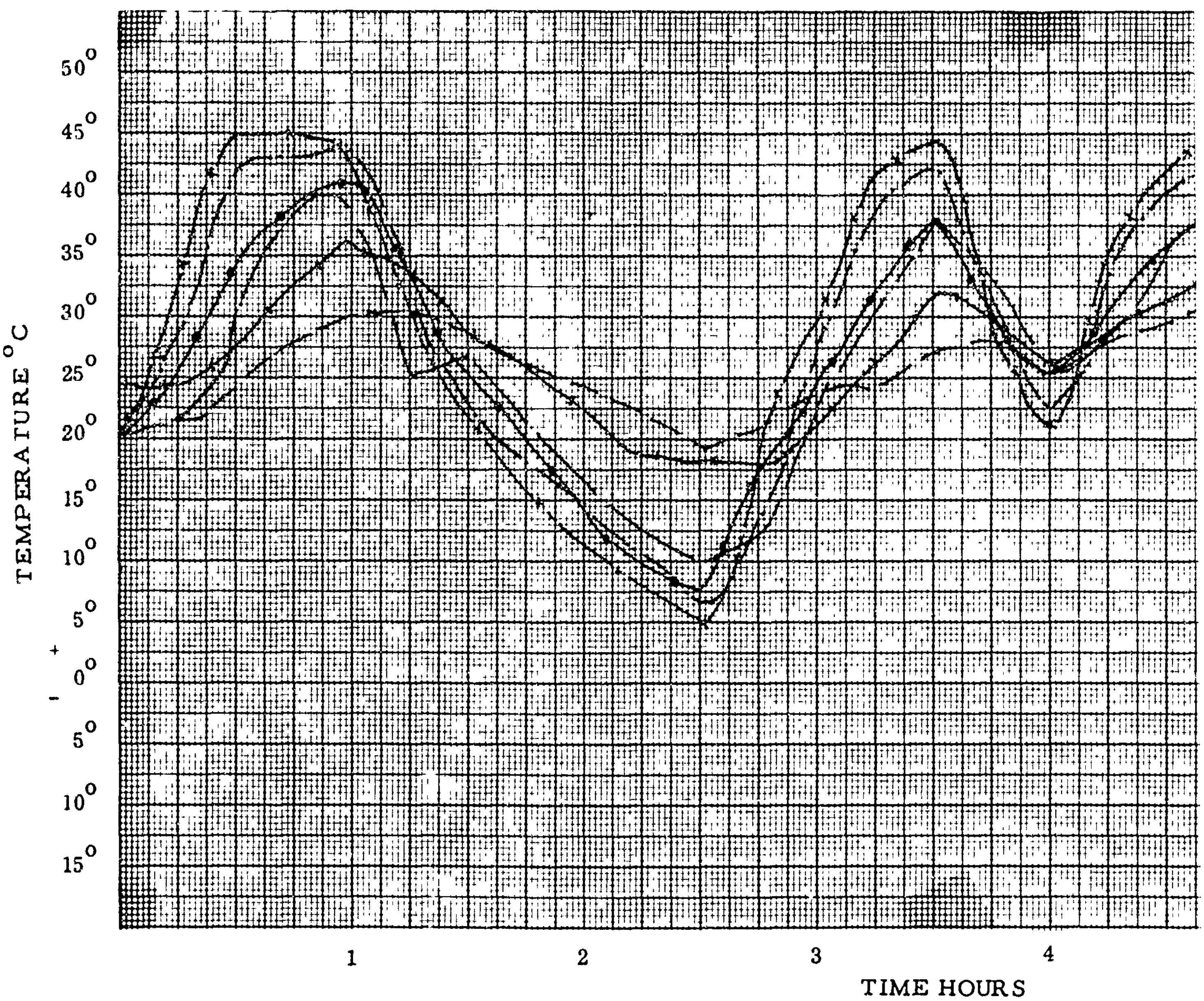
Figure 3-84. Details of Centrifuge Mounting

3.13.4.1.3 Thermal-Vacuum Tests. The thermal-vacuum tests were conducted at General Dynamics, San Diego Division, San Diego, California during the period of 26 May 1961 to 31 May 1961. These tests were conducted using heating lamps and liquid nitrogen in a vacuum of approximately 1.5×10^{-5} mm of Hg. Thermal-vacuum tests were performed while cycling the temperature between maximum values in 100-minute periods. Thermocouples provided graphs of the thermal response at various locations inside and outside the Spacecraft. The graphs showed the desired thermal lag between the instrumentation compartment and the outer surface of the Spacecraft. Throughout these tests the Spacecraft was operational, and Cubic collected data on the operation of the payload. (Refer to appendix A for tables containing this data.)

3.13.4.1.3.1 Thermal shock tests were performed by soaking the Spacecraft at near liquid nitrogen temperature for three hours. This brought the temperature inside the Spacecraft down to approximately -10°C. The thermal cycling was begun from this point with the expected results which further verified the thermal design.

3.13.4.1.3.2 The only difficulty experienced as a result of the entire environmental test program was intermittent operation of one of the telemetry circuits after the thermal shock test. Corrective action was taken by enlarging the contact area around the eyelets on the printed circuit boards. The board redesign work is complete and new printed circuit boards were fabricated with the improved features. No other difficulties were experienced and the test results are considered completely satisfactory.

3.13.4.1.3.3 Notes derived from the actual test procedure are presented in the following paragraphs. The procedure references the sheets of figure 3-85 containing the respective graphs of the thermocouple responses. Table XVII is a list of the thermocouples and their location on the Spacecraft. Figure 3-86 shows the heat lamp distribution utilized, and figure 3-87 indicates the type of liquid nitrogen cooling manifold in the General Dynamics tank. Figure 3-88 shows the Spacecraft



START OF RUN 1. 17:15 5/27/61

LEGEND

$1^{\circ}C$ = 2 MM
1 HOUR = 4 CM

INSIDE SKIN
INSIDE SOLAR CELL
CENTER TUBE TOP
CENTER TUBE BOTTOM
OUTER SKIN OPPOSITE FLIGHT PLUG
OUTSIDE SOLAR CELL

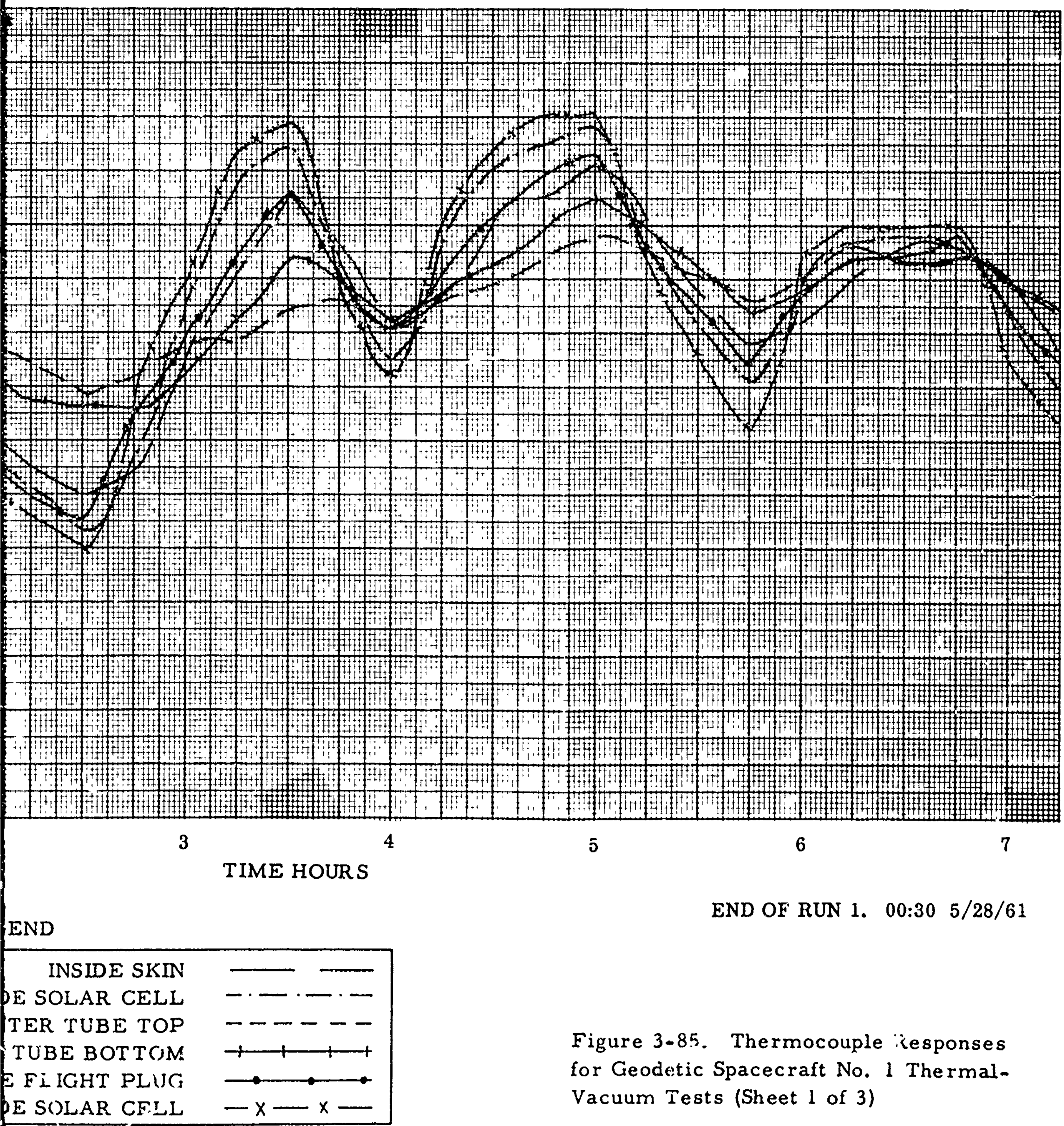
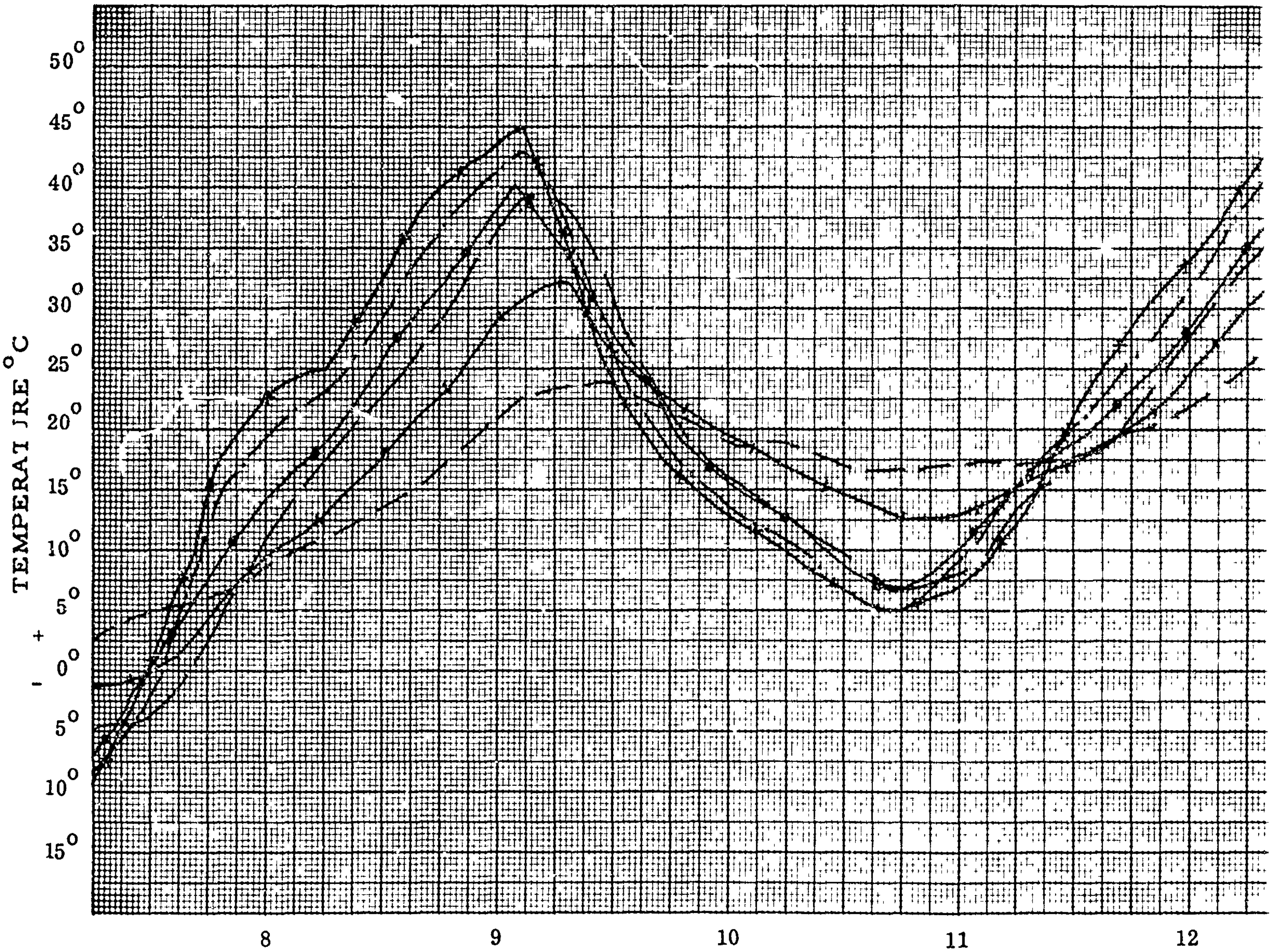


Figure 3-85. Thermocouple Responses for Geodetic Spacecraft No. 1 Thermal-Vacuum Tests (Sheet 1 of 3)

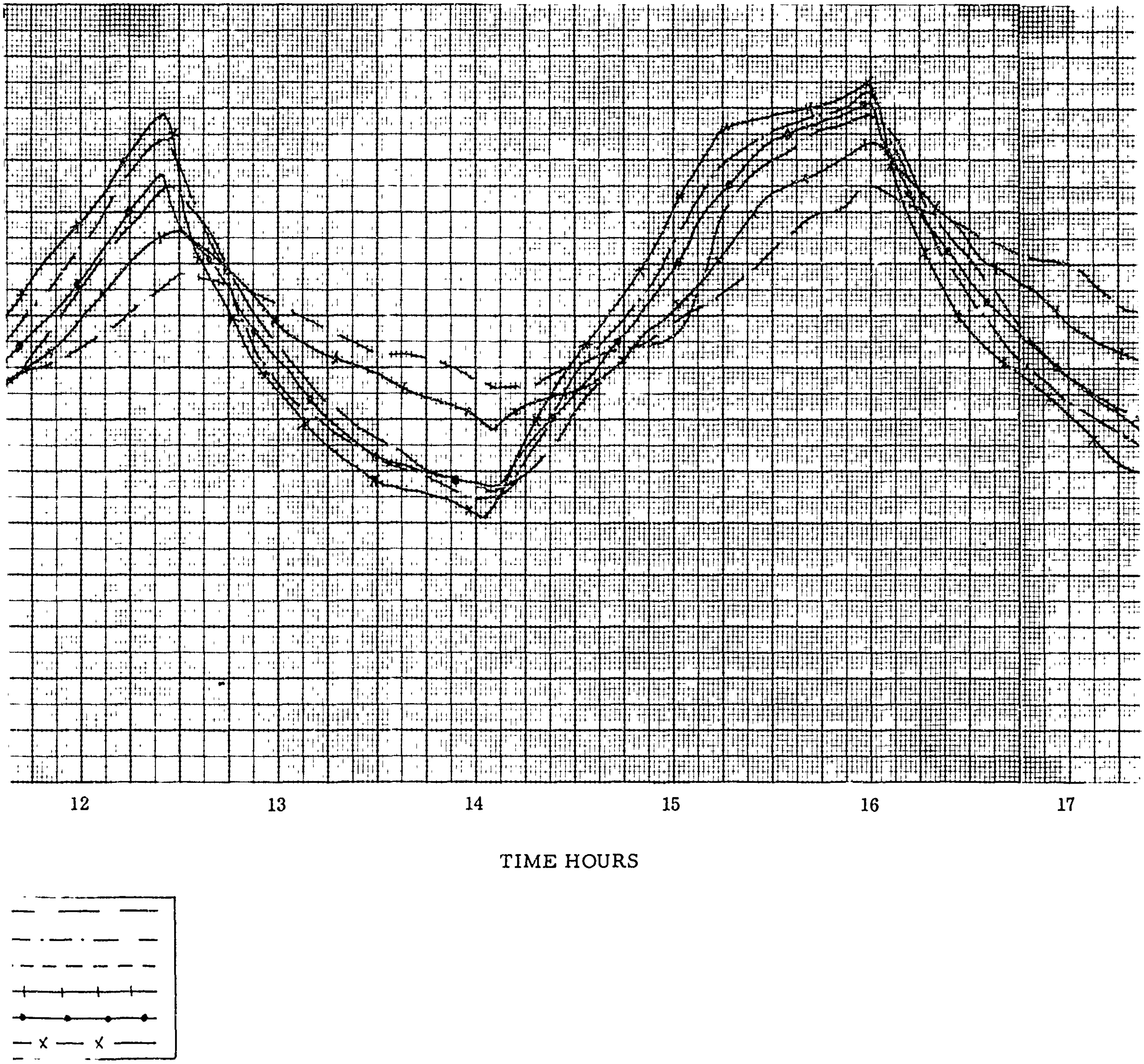


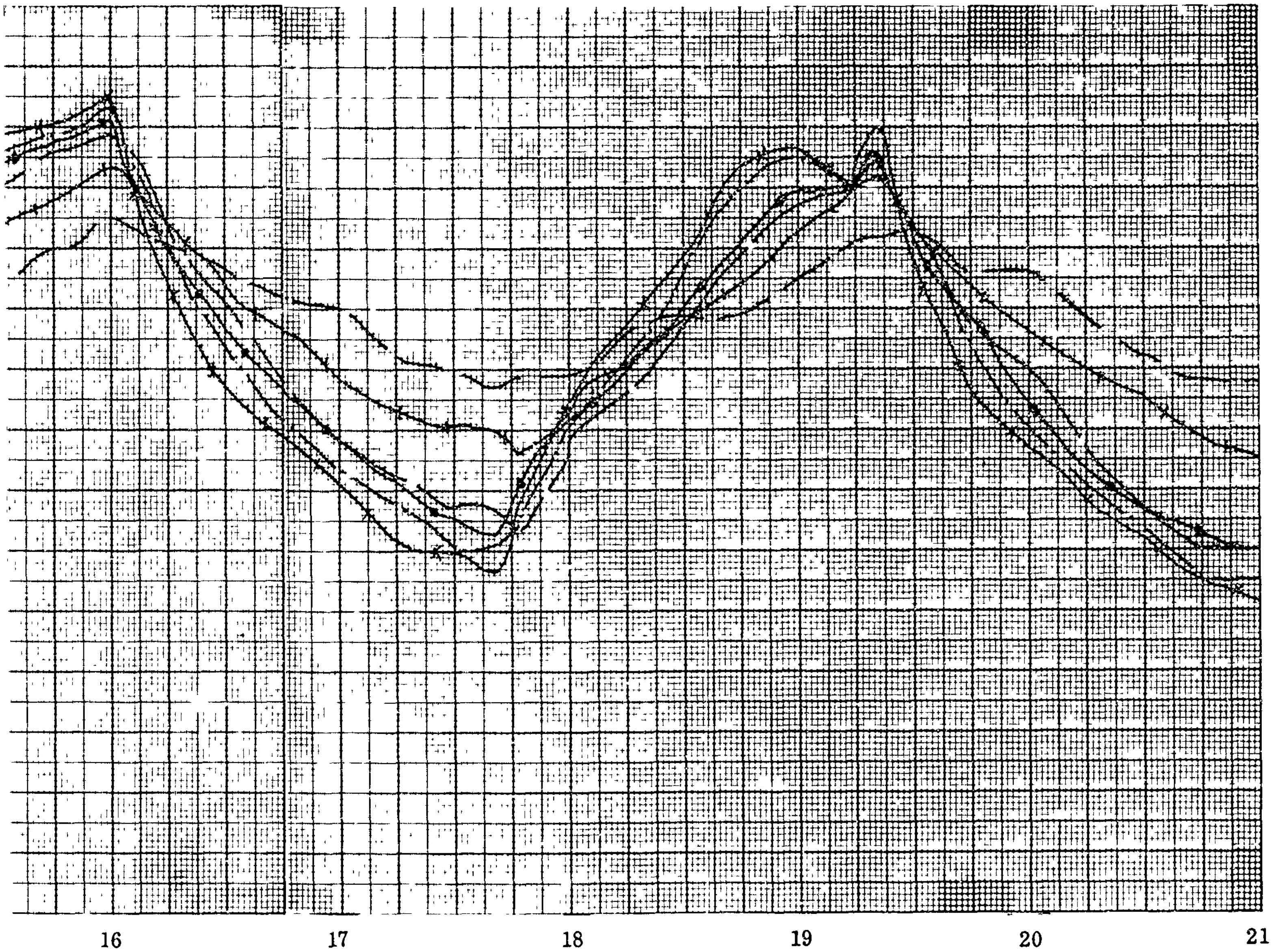
START OF RUN 2. 10:15 5/29/61

$1^{\circ}C$ = 2 MM
1 HOUR = 4 CM

LEGEND

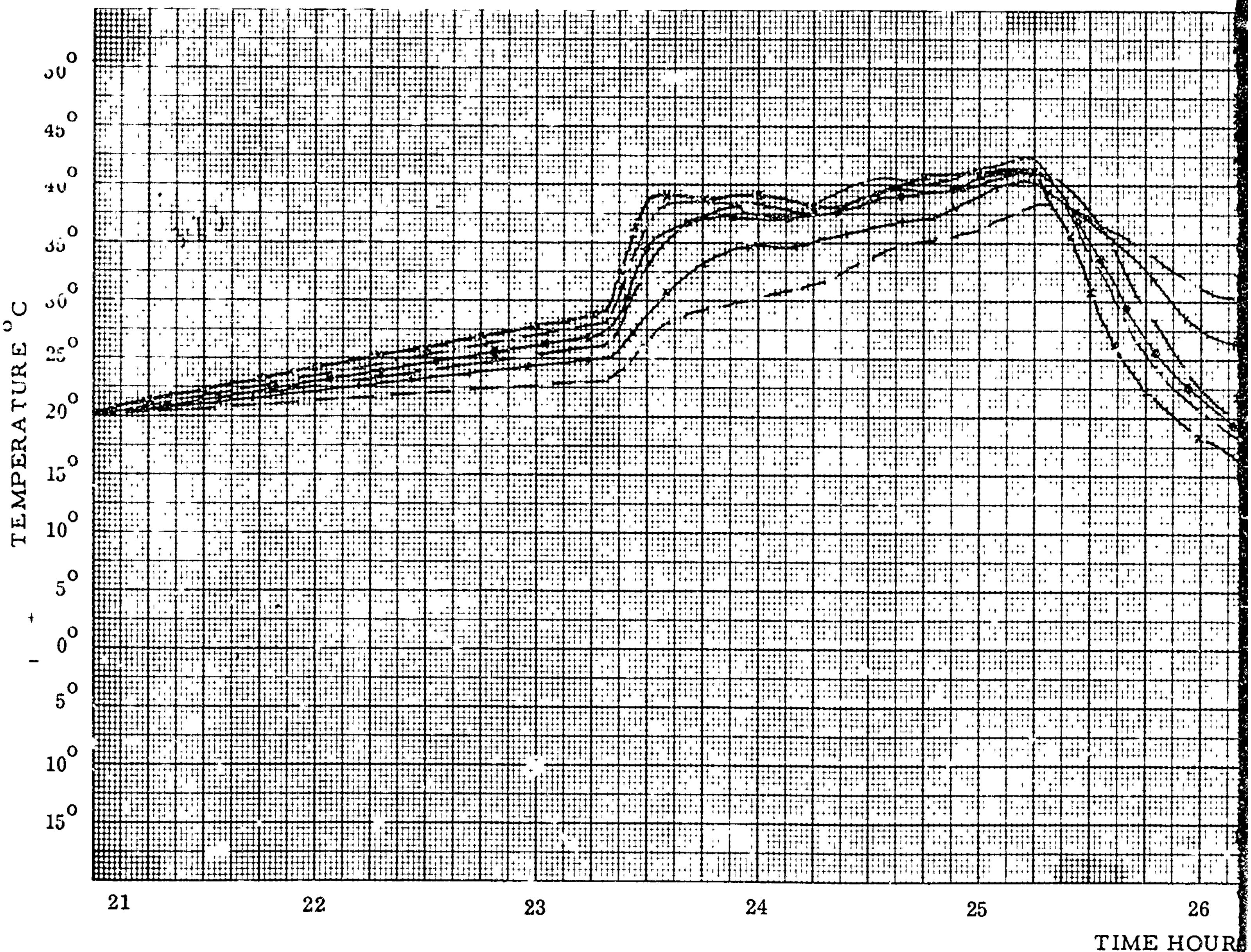
INSIDE SKIN	— — —
INSIDE SOLAR CELL	— · — · — ·
CENTER TUBE TOP	– – – – – –
CENTER TUBE BOTTOM	—+—+—+—
OUTSIDE SKIN OPPOSITE FLIGHT PLUG	—•—•—•—
OUTSIDE SOLAR CELL	— x — x —


12
13
14
15
16
17
TIME HOURS



END OF RUN 2. 24:00 5/29/61

Figure 3-85. Thermocouple Responses for Geodetic Spacecraft No. 1 Thermal-Vacuum Tests (Sheet 2 of 3)



START OF RUN 3. 07:15 5/31/61

1°C = 2 MM
1 HOUR = 4 CM

LEGEND

INSIDE SKIN
INSIDE SOLAR CELL
CENTER TUBE TOP
CENTER TUBE BOTTOM
OUTER SKIN OPPOSITE FLIGHT PLUG
OUTSIDE SOLAR CELL

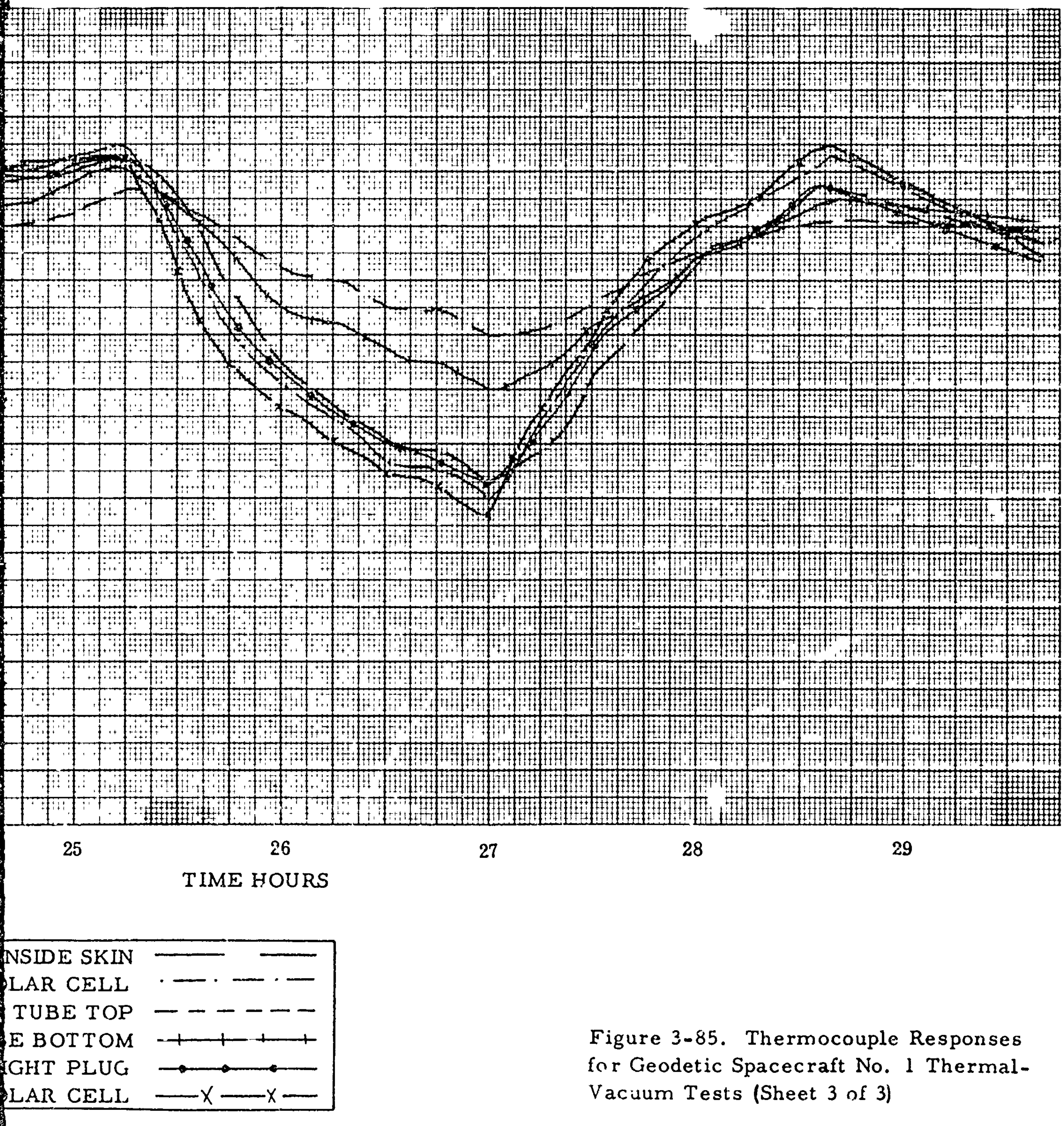


Figure 3-85. Thermocouple Responses for Geodetic Spacecraft No. 1 Thermal-Vacuum Tests (Sheet 3 of 3)

TABLE XVII

THERMOCOUPLE IDENTIFICATION
FOR
THERMAL-VACUUM TEST OF GEODETIC SPACECRAFT NO. 1

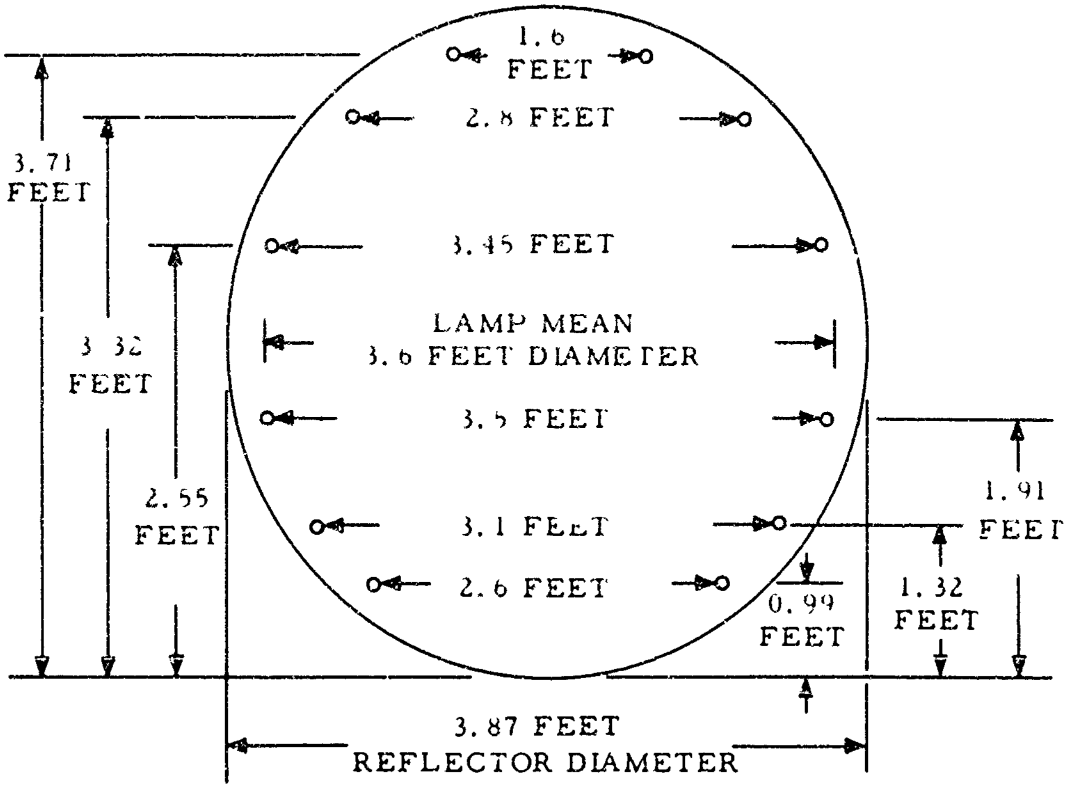
No	Location on the Specimen
1	Multiplier Temperature
2	Transmitter Temperature
3	Inside Skin Temperature
4	Inside Solar Cell Temperature
5	Center Tube Top Temperature
6	Center Tube Bottom Temperature
7	Outer Skin (Opposite Flight Plug) Temperature
8	Solar Cell Outside Temperature
NOTE: All thermocouples were copper-constantan	

installed in the test environment Table XVIII lists data on the infrared lamps. The test was conducted in three test steps as follows:

(1) Test step No. 1 (see figure 3-85, sheet 1)

Time	
1715	Chamber evacuated - 5/27/61
1715	Measured ambient conditions - 5/27/61
1715	Started heat cycle, maximum temperature on thermocouple No. 8 to be 45°C

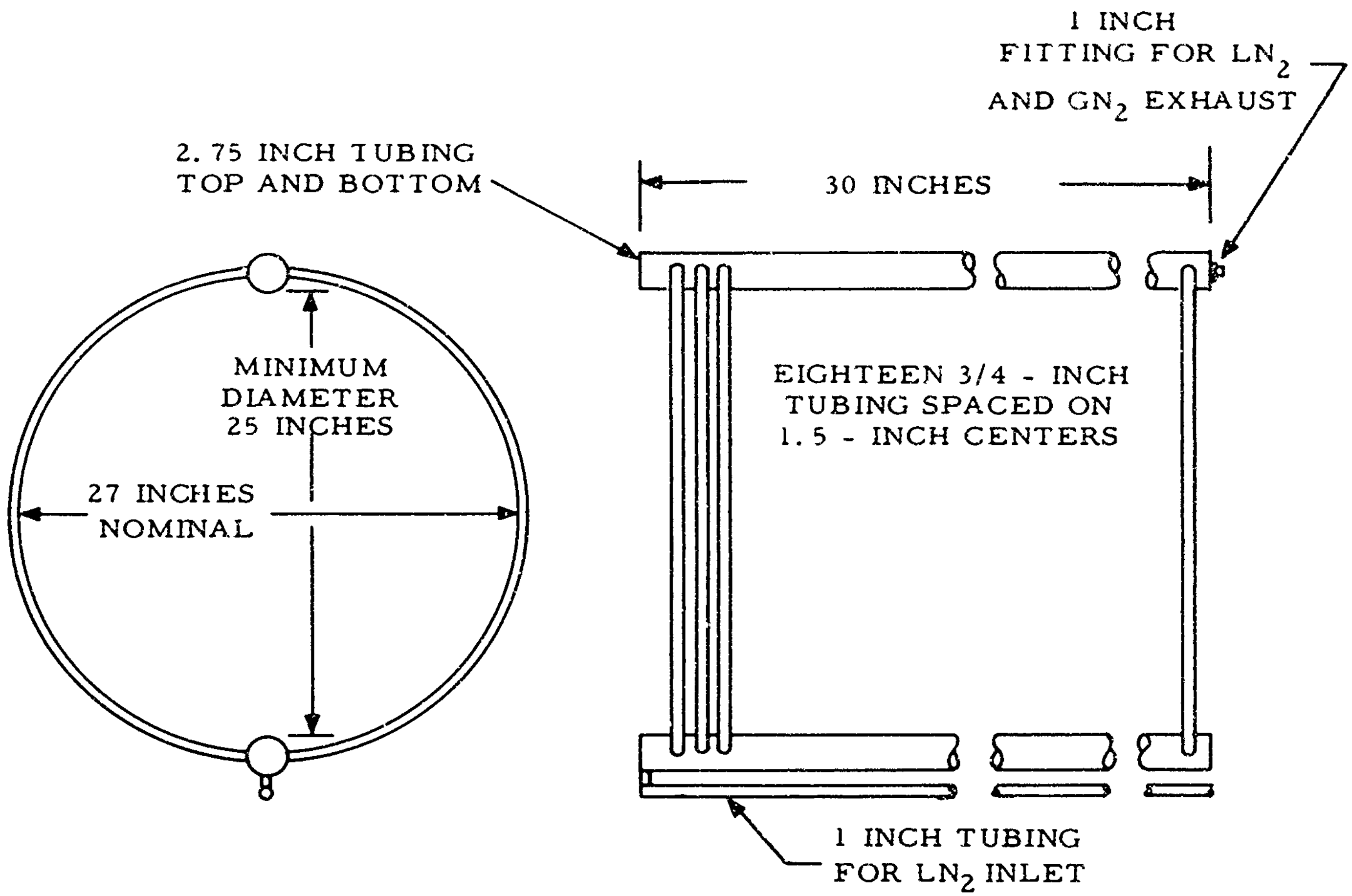
HEAT LAMP DISTRIBUTION



NOTES:

1. REFLECTOR MATERIAL -- ALUMINUM.
2. REFLECTOR FINISH (INSIDE) -- DULL - NOT POLISHED.
3. LIGHTED LENGTH OF LAMPS -- 25 INCHES (2.08 FEET).
4. LAMP CENTERLINE TO REFLECTOR SURFACE -- 0.27 FEET.
5. TYPE OF LAMPS -- INFRARED, TUBULAR QUARTZ G.E. 2500T3 (460V-500V).
6. REFLECTOR COVERS FULL LIGHTED LENGTH OF LAMP.

Figure 3-86. Heat Lamp Distribution Utilized for Thermal-Vacuum Tests



NOTE: ALL TUBING OF STAINLESS STEEL. ALL JOINTS HELIARCED.

Figure 3-87. Liquid Nitrogen Cooling Manifold in General Dynamics Vacuum Tank

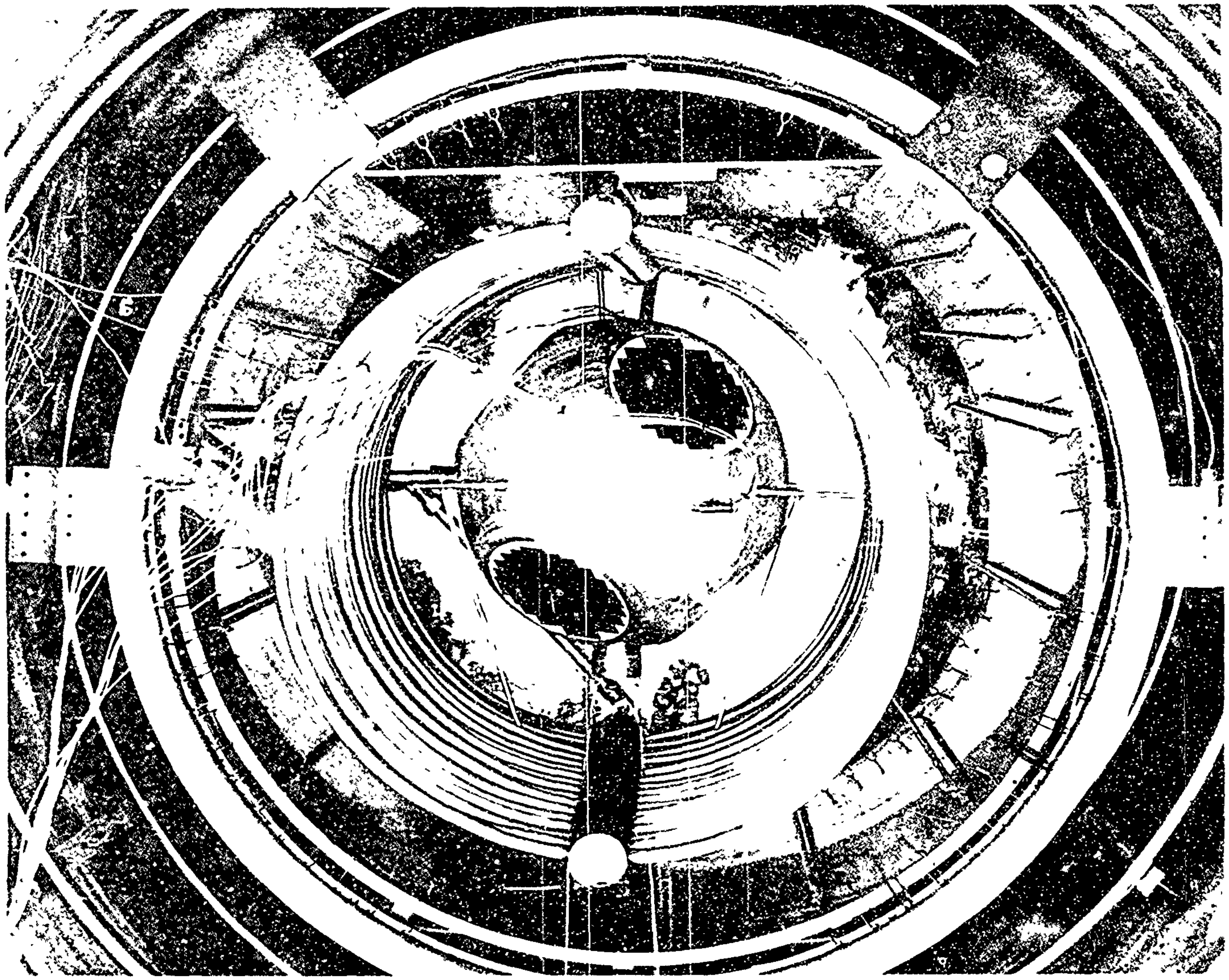


Figure 3-88. Geodetic Spacecraft Mounted in Thermal-Vacuum Tank

TABLE XVIII

INFRARED LAMP DATA

G. E. 2500T3 Tubular Quartz
(This data is based on one lamp only)

Volts (RMS)	Amperes	Watts
20.0	1.1	22.0
30.0	1.3	39.0
40.0	1.5	60.0
50.0	1.65	82.5
60.0	1.70	102.0
70.0	1.85	129.5
80.0	1.95	156.0
90.0	2.10	189.0
100.0	2.20	220.0
110.0	2.35	258.5
120.0	2.40	288.0
125.0	2.48	310.0
140.0	2.65	371.0
145.0	2.70	391.5

All voltage and current values measured with 3/4 per cent Weston meters.

Time	
1813	Heat was on for approximately 45 minutes. Heat off at 1813, cold started
	To remain cold until thermocouple No. 8 reaches $5^{o}C$
1950	Heat on (80 volts across heater) while liquid nitrogen was still in the cooler for 15 minutes, then cold off
2045	Heat off, cold started
2115	Heat on, cold off - 80 volts across heaters
2215	Heat off, cold on
	On the next heat cycle, limit thermocouple No. 8 to $35^{o}C$ instead of $45^{o}C$ (per Varson)
2245	Cold cycle extended 15 minutes (per Varson)
2300	Heat on, cold off
2400	Heat off, cold on
0030	5/28/61 - Heat off, cold off, chamber remaining evacuated. End of test step No. 1

(2) Test step No. 2 (see figure 3-85, sheet 2)

Time	
0730 to 1015	5/29/61 - Chamber evacuated and temperature reduced to as cold as possible
1025	Heat on - Adjusted heater voltage to get a rate of rise in temperature on thermocouple No. 8 of $0.5^{o}C$ per minute. Actually got $0.475^{o}C$ per minute with a heater voltage of 40 volts.

Time	
1205	Heat off - Total heat cycle 100 minutes - cold started
1345	Heat on, heater voltage maximum set at 56 volts - Cold off
1415	Heater voltage adjusted to 53 volts
1430	Heater voltage adjusted to 55 volts
1500	Heater voltage adjusted to 57 volts
1525	Heat off - Cold started
1705	Heat on - Cold off. Heater voltage set for maximum 64 volts Monitor thermocouple No. 8; when it reaches 45°C stabilize for approximately 5 minutes, then increase the temperature until couple No. 8 reaches 55°C or until thermocouple No. 6 reaches at least 42°C whichever occurs first.
1715	Heater voltage set for maximum 65 volts
1745	Heater voltage set for maximum 67.5 volts
1747	Heater voltage set for maximum 70 volts
1845	Increased duration of heat cycle 15 minutes to attain 42°C on thermocouple No. 6
1900	Heat off, cold started
1908	Cubic reported telemetry erratic, then went dead during temperature reading
2040	Heat on - Heater voltage 68 volts - cold off

Time	
2055	Cubic reported telemetry began to function
2220	Maximum heat and maximum cold applied simultaneously for 3 minutes, heater voltage 120 volts. After 3 minutes, heat off, cold remained on
2300	Cubic reported telemetry still functioning
2315	Cubic reported telemetry went dead
2400	Cold off, heat off, chamber remaining evacuated. End of step No. 2

(3) Test step No. 3 (see figure 3-85, sheet 3)

Time	
0715	5/31/61 - Recorded ambient conditions and began to re-establish altitude
0930	Photographed the test specimen through the chamber window using heaters for light at full power (120 volts). Heaters on for 2 minutes each photograph. Two photographs made 5 minutes apart. Photos to be 4" x 5" color. Raise the temperature of thermocouple No. 8 to 40°C in 100 minutes. After reaching 40°C, hold this temperature or increase the temperature whichever is required to raise the temperature of thermocouple No. 6 to 40°C. After No. 6 reaches 40°C stabilize No. 6 at 40°C for 15 minutes (per Varson and Sharman).
0935	Heat on
0945	Heat reduced to slow the rate of temperature increase Use couple No. 5 instead of No. 6 to stabilize on at 40°C - Ref second paragraph, 0930

Time	
1000	Couple No. 8 at 38.5°C stabilizing for the remainder of the heat cycle - Heat continuously adjusted to maintain 40°C on couple No. 8.
1115	Heat time extended 15 minutes to stabilize couples No. 5 and No. 6 at 40°C
1130	Heat off, started cold, Cubic reports telemetry is functioning
1315	Heat on, cold off
1455	Heat off, chamber walls being cooled with water, altitude being reduced
1556	End of test program

3.13.4.2 Description of Tests for Spacecraft No. 2. Geodetic Spacecraft No. 2 has successfully completed the environmental test program. This consisted of random and sinusoidal vibration tests, acceleration tests, and thermal-vacuum tests.

3.13.4.2.1 Random and Sinusoidal Vibration Tests. The vibration tests discussed in the following paragraphs were performed at Ryan Electronics, San Diego, California on 4 August 1961. The Spacecraft was tested to specification in the three major axes. Refer to table XVI for a list of the test equipment utilized. See figure 3-75 for the test setup. The Spacecraft was nonoperational during this test. The procedure was as follows:

(1) The test specimen was secured to a test fixture and was mounted to a vibration exciter in a position to apply vibration along the first transverse axis. The frequency of vibration was then scanned from 5 to 50 cps at ±0.6g in 1.66 minutes, 50 to 500 cps at ±1.4g in 1.66 minutes; 500 to 2000 cps at ±2.8g in 1 minute; and from 2000 to 3000 cps at ±11.3g or to the limits of the vibration machine in 0.6 minute. The frequency of vibration was swept at an approximate logarithmic rate.

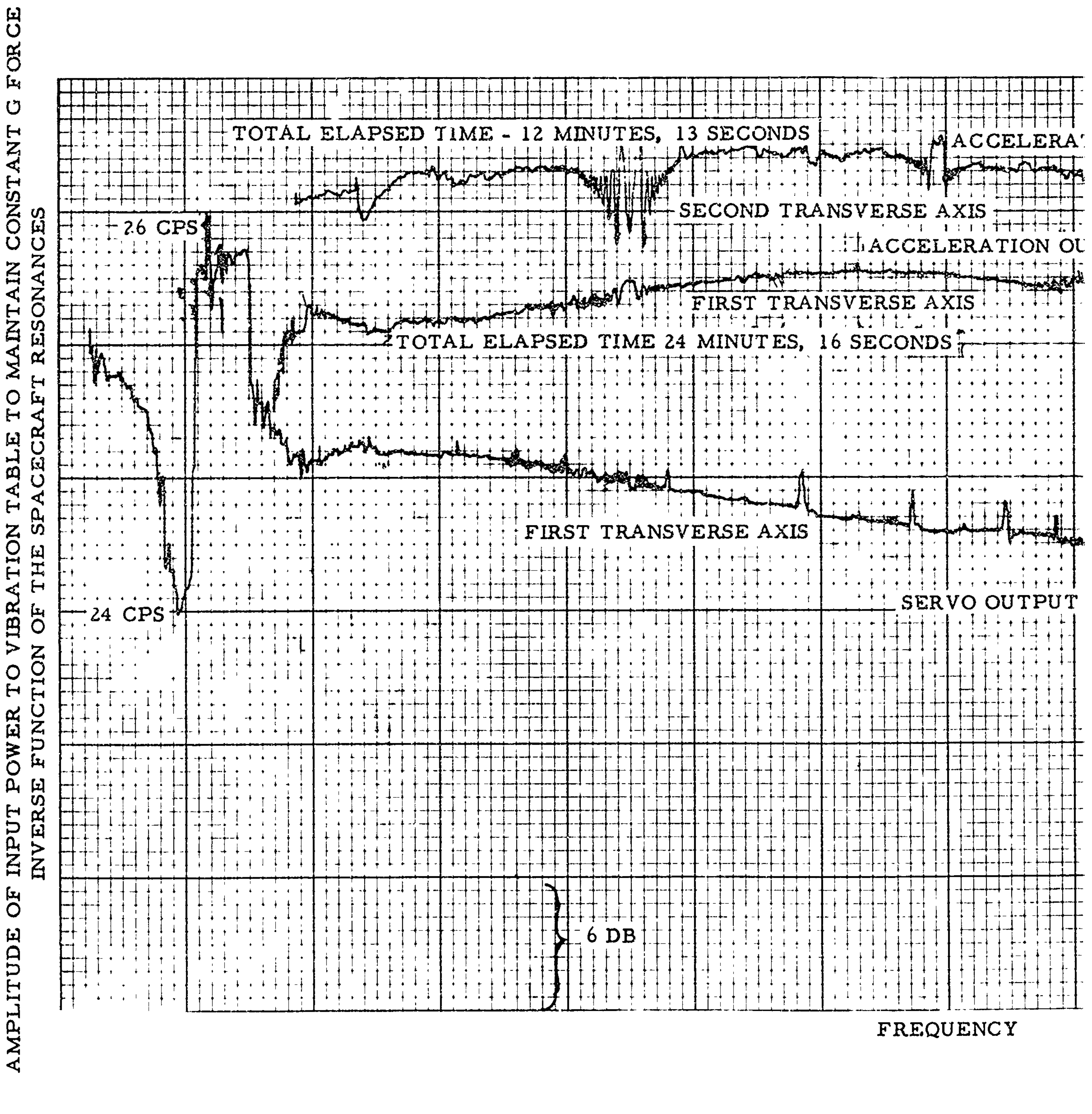
(2) With the test specimen still mounted in the first transverse axis, the vibration exciter response was equalized to within ±2 db of a flat power spectral density curve within the frequency band of 30 to 2000 cps. The acceleration clipping was adjusted so that 99.7 per cent of the true Gaussian distribution was realized and displacement clipping was adjusted to 0.5 inch D. A. The test specimen was then subjected to random vibration with a frequency band of 30 to 2000 cps at a 3.7g (rms) level for 4 minutes.

(3) Paragraphs (1) and (2) above were then repeated in the second transverse axis.

(4) The test specimen and test fixture were then mounted on the vibration exciter in a position to apply vibration along the longitudinal axis. The frequency of vibration was then scanned from 5 to 50 cps at 0.95 inch D. A. or ±1.5g (whichever limited in 1.66 minutes); 50 to 500 cps at ±7.3g in 1.66 minutes; 500 to 2000 cps at ±14g in 1 minute; and from 2000 to 3000 cps at ±36g or to the limits of the vibration machine in 0.5 minute. The frequency of vibration was swept at an approximate logarithmic rate.

(5) With the test specimen still mounted in the longitudinal axis, the vibration exciter response was equalized to within ±2 db of a flat power spectral density curve within the frequency band of 30 to 2000 cps. The acceleration clipping was adjusted so that 99.7 per cent of the true Gaussian distribution was realized and displacement clipping was adjusted to 0.5 inch D. A. The test specimen was then subjected to random vibration with a frequency band of 30 to 2000 cps at a 3.7g (rms) level for 4 minutes. The vibration test was completed as described in the test procedure. During and at the conclusion of the vibration test, the test specimen was examined for mechanical damage. As no evidence of mechanical damage was apparent, it was concluded that the Spacecraft had successfully completed the vibration testing. See figure 3-89 for a plot of the resonances observed in the transverse axis, and figure 3-90 for resonances in the longitudinal axis.

3.13.4.2.2 Acceleration Tests. Acceleration tests were performed at General Dynamics, San Diego Division, San Diego, California on 7 August 1961. The Spacecraft successfully endured 20 g acceleration in the thrust axis with 1 g in



AMPLITUDE OF INPUT POWER TO VIBRATION TABLE TO MAINTAIN CONSTANT G FORCE
INVERSE FUNCTION OF THE SPACECRAFT RESONANCES
TOTAL ELAPSED TIME - 12 MINUTES, 13 SECONDS
ACCELERA
SECOND TRANSVERSE AXIS
26 CPS
ACCELERATION OU
FIRST TRANSVERSE AXIS
TOTAL ELAPSED TIME 24 MINUTES, 16 SECONDS
FIRST TRANSVERSE AXIS
24 CPS
SERVO OUTPUT
6 DB
FREQUENCY

A

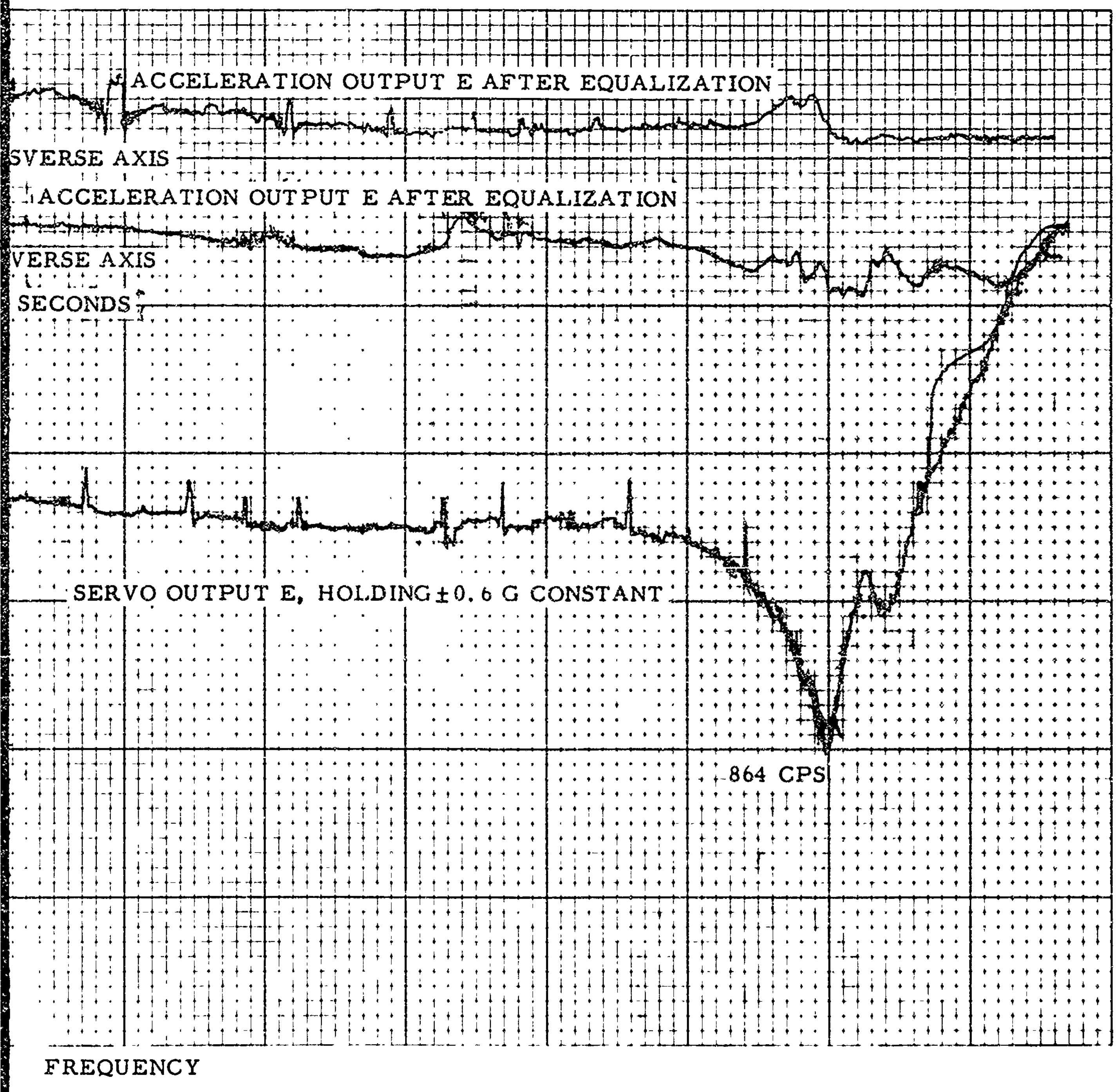
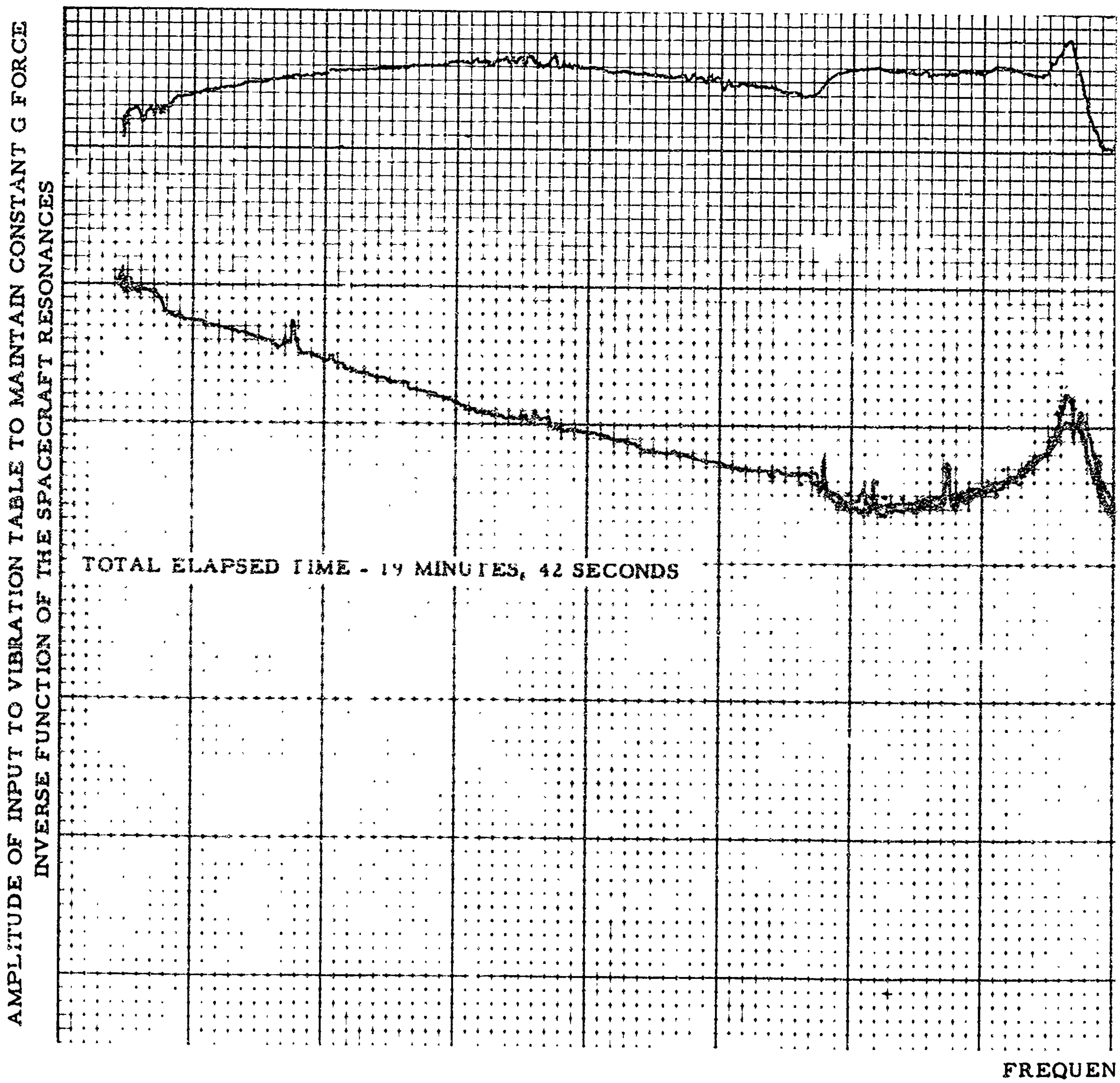


Figure 3-89. Spacecraft No. 2 Resonant Responses, Transverse Axis

B


AMPLITUDE OF INPUT TO VIBRATION TABLE TO MAINTAIN CONSTANT G FORCE
INVERSE FUNCTION OF THE SPACECRAFT RESONANCES
TOTAL ELAPSED TIME - 19 MINUTES, 42 SECONDS
FREQUEN

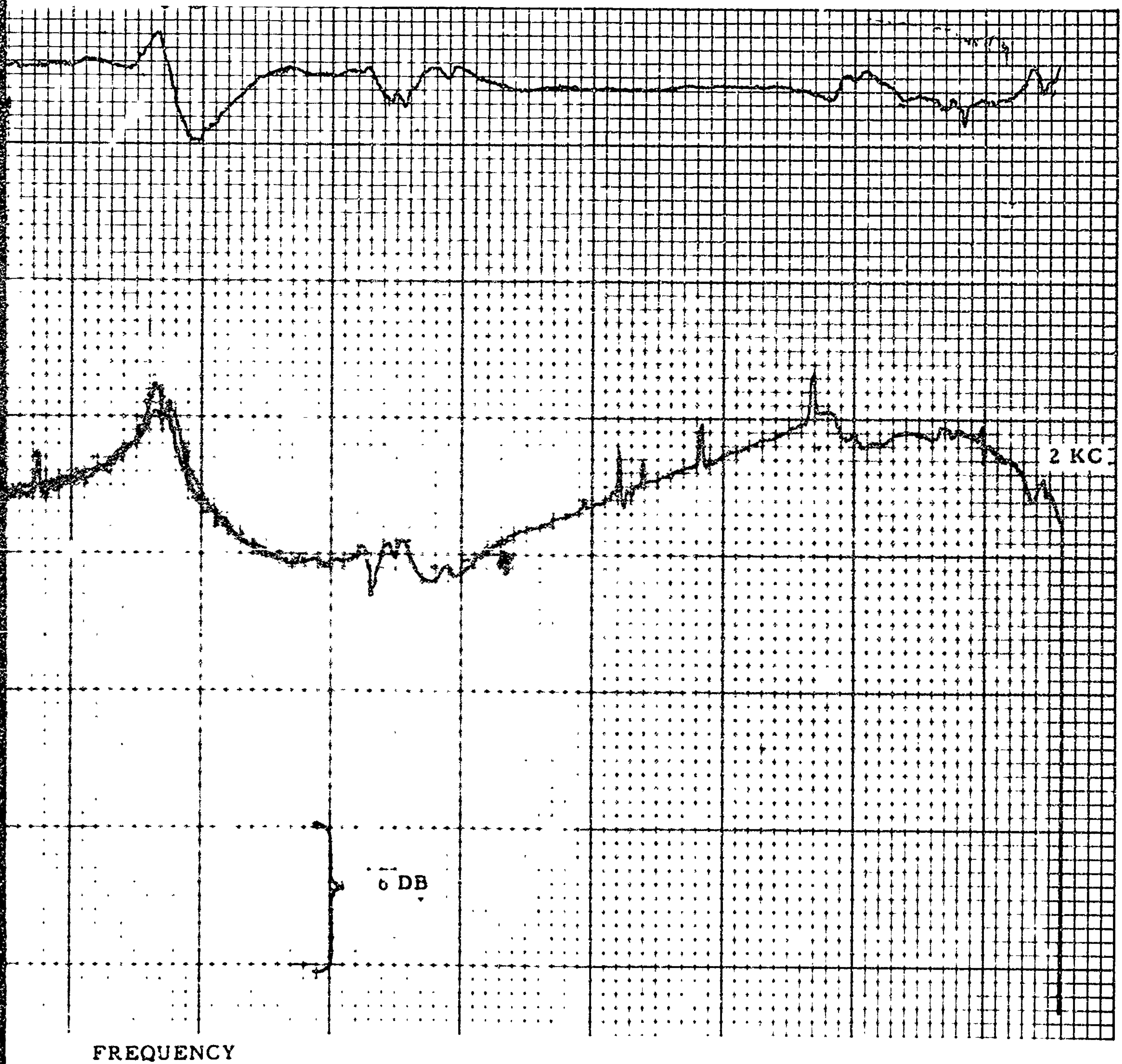


Figure 3-90. Spacecraft No. 2 Resonant Responses, Longitudinal Axis

the transverse axis for a three-minute period, and 35g for a one-minute period.

3.13.4.2.3 Thermal-Vacuum Tests. The thermal-vacuum tests for Spacecraft No. 2 were conducted at General Dynamics, San Diego Division, San Diego, California during the period of 7 August 1961 to 11 August 1961. These tests were conducted using heating lamps and liquid nitrogen in a vacuum of approximately 1.5×10^{-6} mm of Hg. Thermal-vacuum tests were performed while cycling the temperature between -20^{o}C and +55^{o}C in 100-minute periods. Thermocouples provided graphs of the thermal response at various locations inside and outside the Spacecraft. The graphs showed the desired thermal lag between the instrumentation compartment and the outer surface of the Spacecraft. Throughout these tests the Spacecraft was operational, and Cubic collected data on the operation of the payload. (Refer to appendix B for tables containing this data.) The thermal-vacuum tests were similar to the ones conducted on Spacecraft No. 1, and were performed in thirteen test steps. Table XIX lists the thermocouples utilized to collect the data. Thermocouple response curves for Spacecraft No. 2 are shown in figures 3-91, 3-92, 3-93, and 3-94.

3.13.4.3 Description of Tests for Spacecraft No. 3. Geodetic Spacecraft No. 3 has successfully completed the environmental test program. This consisted of random and sinusoidal vibration tests, acceleration tests, a/ϵ tests, and thermal-vacuum tests.

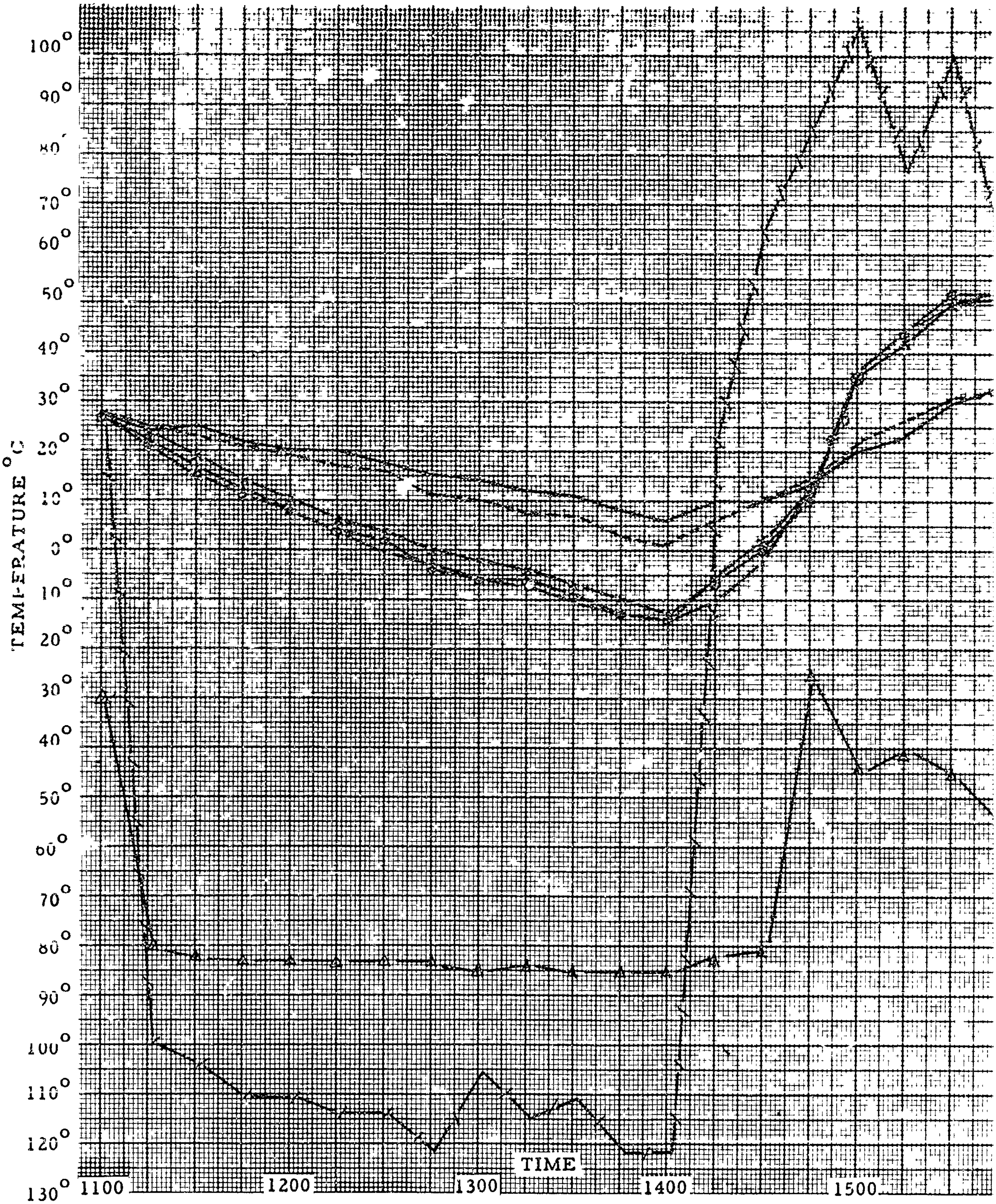
3.13.4.3.1 Random and Sinusoidal Vibration Tests. The vibration tests discussed in the following paragraphs were performed at Ryan Electronics, San Diego, California on 20 July 1961. The Spacecraft was tested to the Scout environment in the three major axes. Both random and sinusoidal vibration tests were performed over the ranges of 20 to 2000 cps, limited only by the capability of the shakers. Refer to table XVI for a list of the test equipment utilized. See figure 3-75 for the test setup. The Spacecraft was nonoperational during this test. The procedure was as follows:

(1) The test specimen was secured to a test fixture and mounted on the vibration exciter in a position to

TABLE XIX

THERMOCOUPLE IDENTIFICATION
FOR
THERMAL-VACUUM TEST OF GEODETIC SPACECRAFT NO. 2

Number	Location on the Specimen
1	Under cover of commutator board
2	Commutator board
3	Center of receiver chassis
4	Base of central tube
5	Central tube between telemeter and transponder
6	Between solar cells (inside)
7	On equator (inside)
8	Center south solar cell (inside)
9	Northern region (inside skin)
10	Center north solar cell (inside)
11	North solar cell (outside)
12	North solar cell (outside)
13	Skin at equator (outside)
14	South solar cell
15	South solar cell
16	Skin
17	Space between cold tubes and Spacecraft
18	Chamber temperature
NOTE: All thermocouples were copper Constantan	


TEMPERATURE °C
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
100°
110°
120°
130°
TIME
1100
1200
1300
1400
1500

DATE	8/8/61
TIME	1100 to 1545
RUN NO.	1
READINGS	20
REF. DATA BOOK	1312

THERMOCOUPLE LEGEND

NO.	LOCATION	REFERENCE
1	UNDER COVER OF COMMUTATOR BOARD	————
4	BASE OF CENTRAL TUBE	- - - - - -
7	ON EQUATOR (INSIDE)	—x—x—
10	CENTER NORTH SOLAR CELL	—o—o—
13	SKIN AT EQUATOR (OUTSIDE)	—·—·—
17	SPACE BETWEEN COLD TUBES AND SPACECRAFT	—/—/—
	CHAMBER PRESSURE, MM Hg	—Δ—Δ—

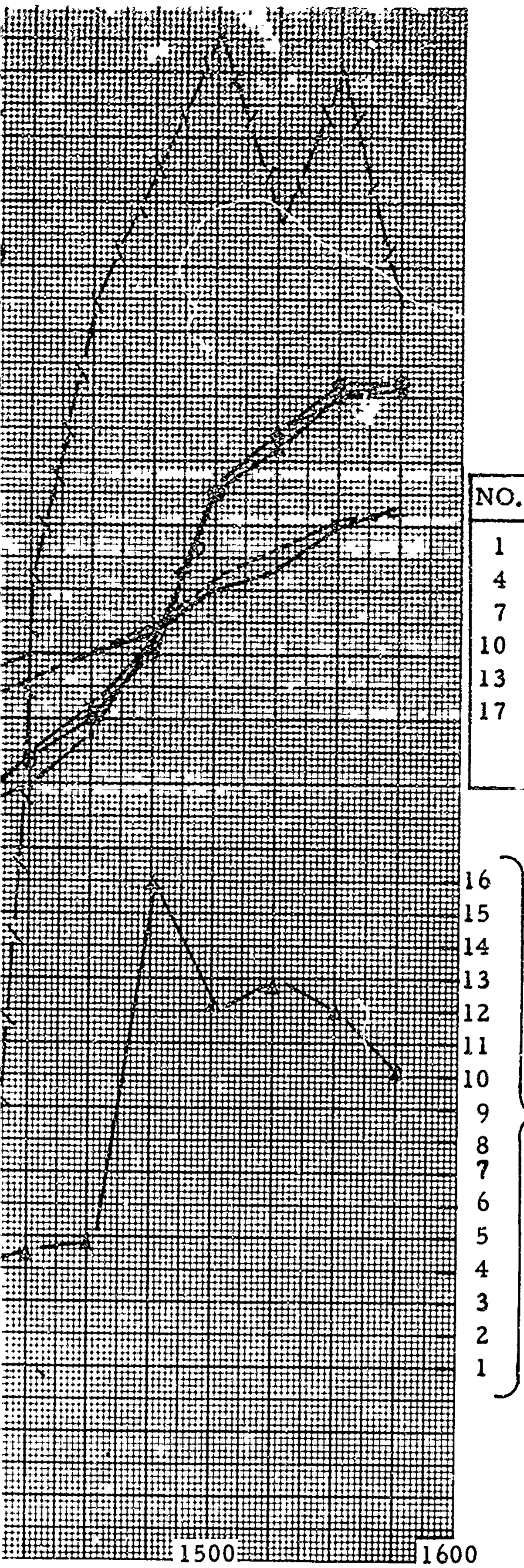
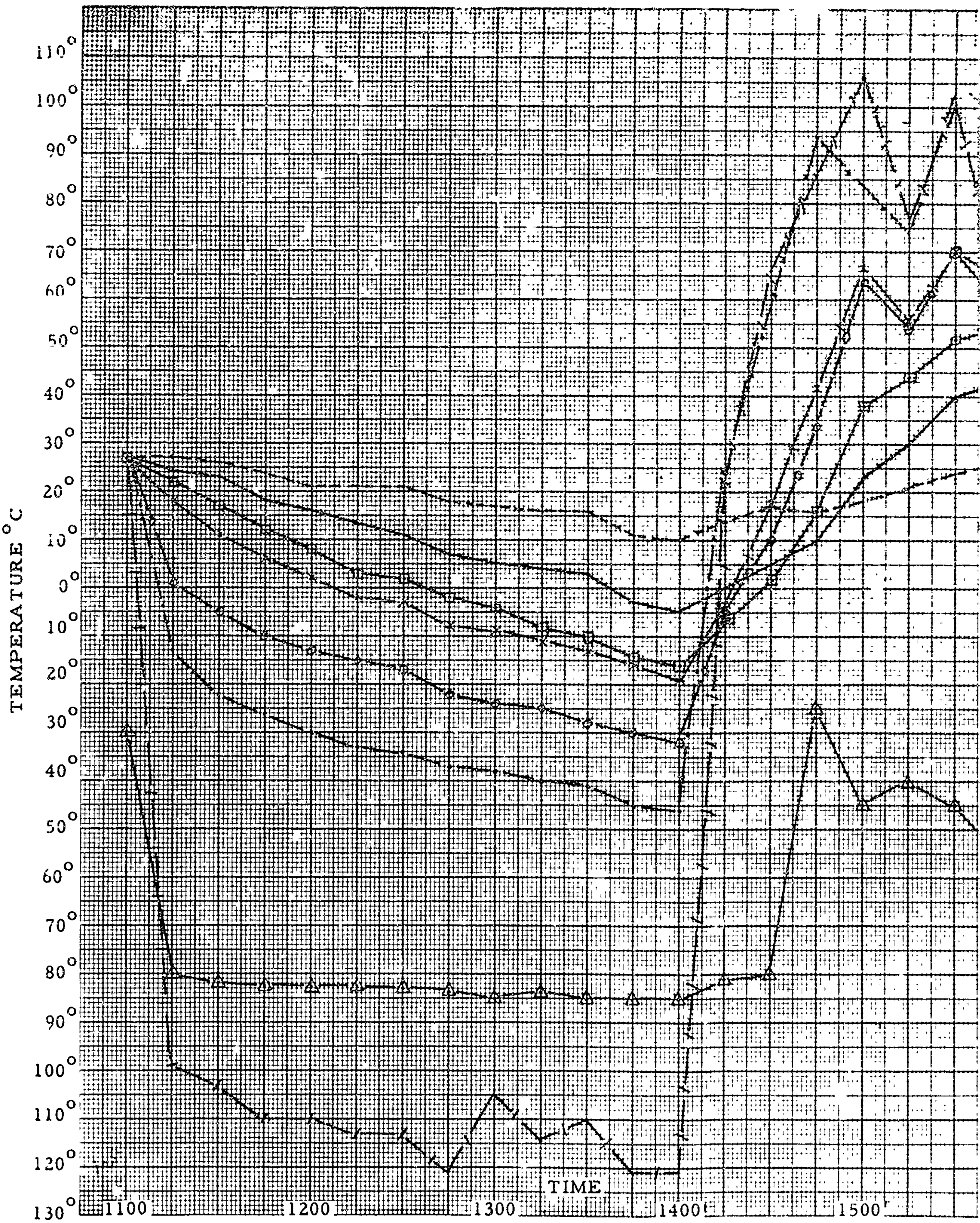


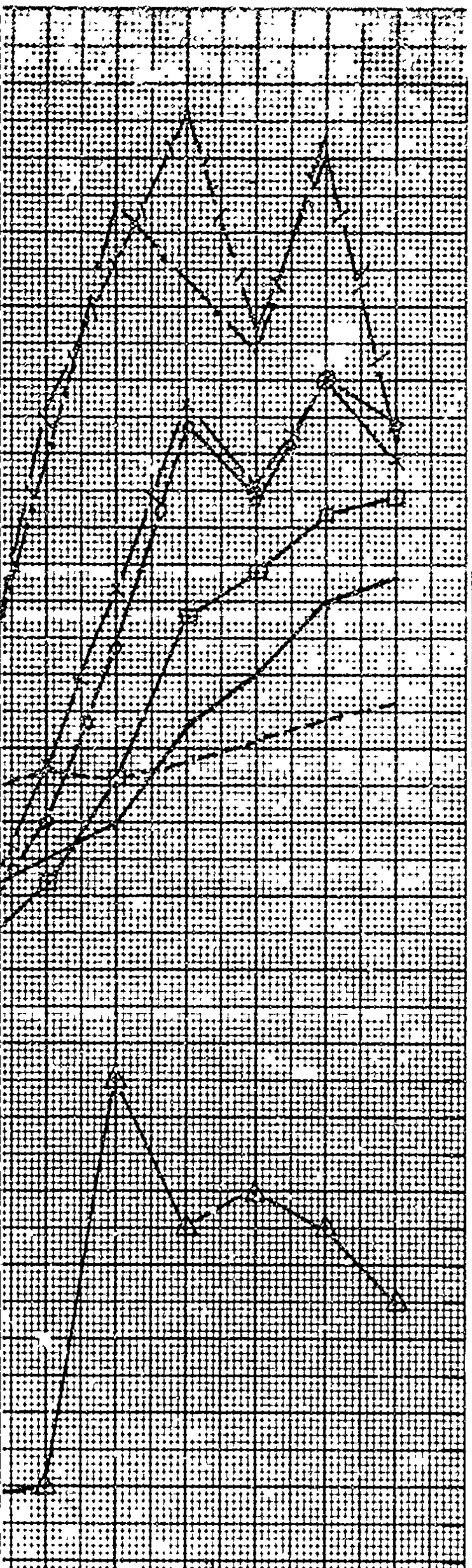
Figure 3-91. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test, Steps 1 and 2 (Sheet 1 of 3)

R.



TEMPERATURE °C
110°
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
100°
110°
120°
130°
TIME
1100
1200
1300
1400
1500

A



DATE	8/8/61
TIME	1100 to 1545
RUN NO.	1
READINGS	20
REF. DATA BOOK	1312

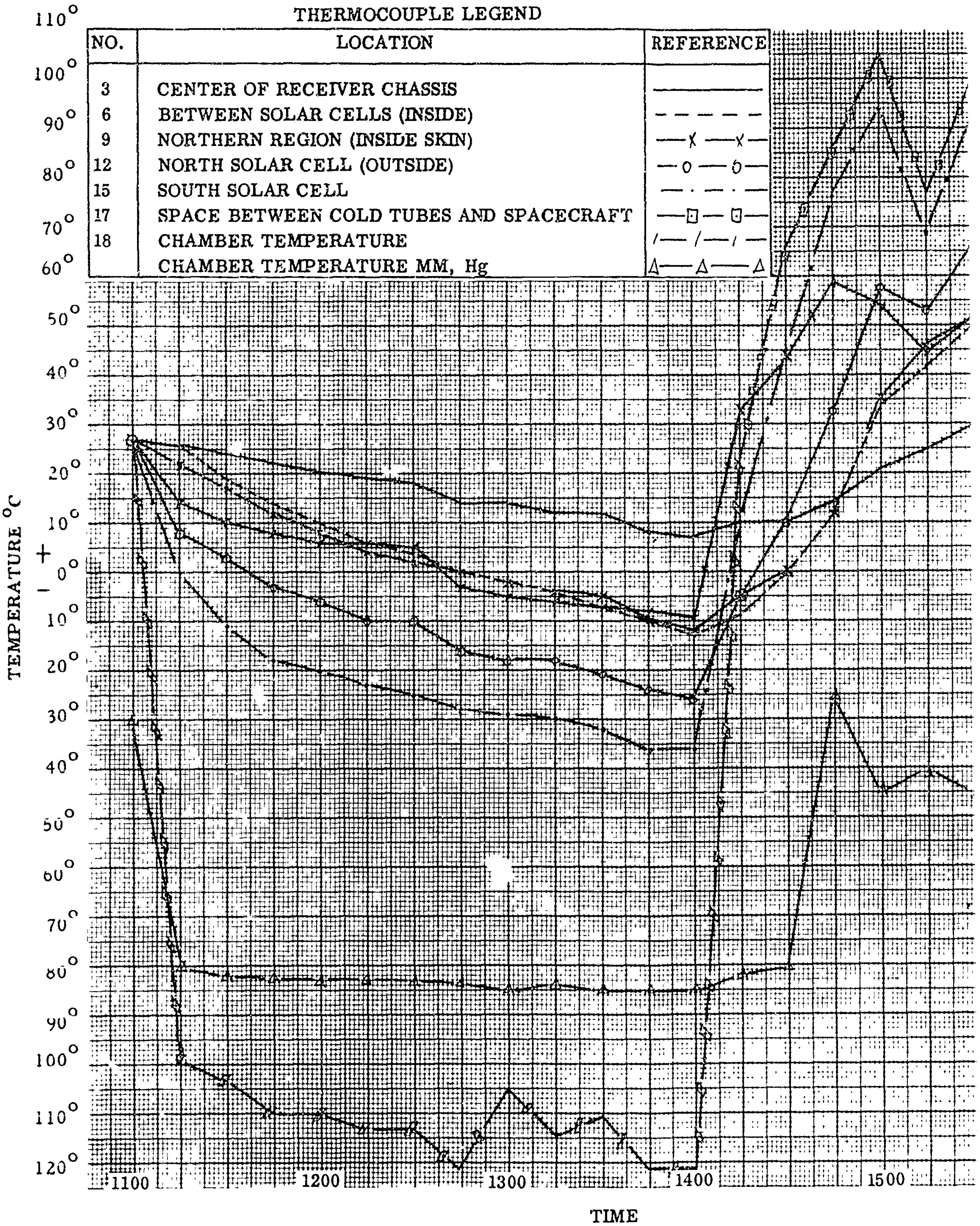
THERMOCOUPLE LEGEND

NO.	LOCATION	REFERENCE
2	COMMUTATOR BOARD	————
5	CENTRAL TUBE BETWEEN TELEMETRY AND TRANSPONDER	— — — —
8	CENTER OF SOUTH SOLAR CELL (INSIDE)	— x — x —
11	NORTH SOLAR CELL (OUTSIDE)	— o — o —
14	SOUTH SOLAR CELL	— · — · —
17	SPACE BETWEEN COLD TUBES AND SPACECRAFT	— / — / —
16	SKIN	— □ — □ —
	CHAMBER PRESSURE, MM Hg	— △ — △ —

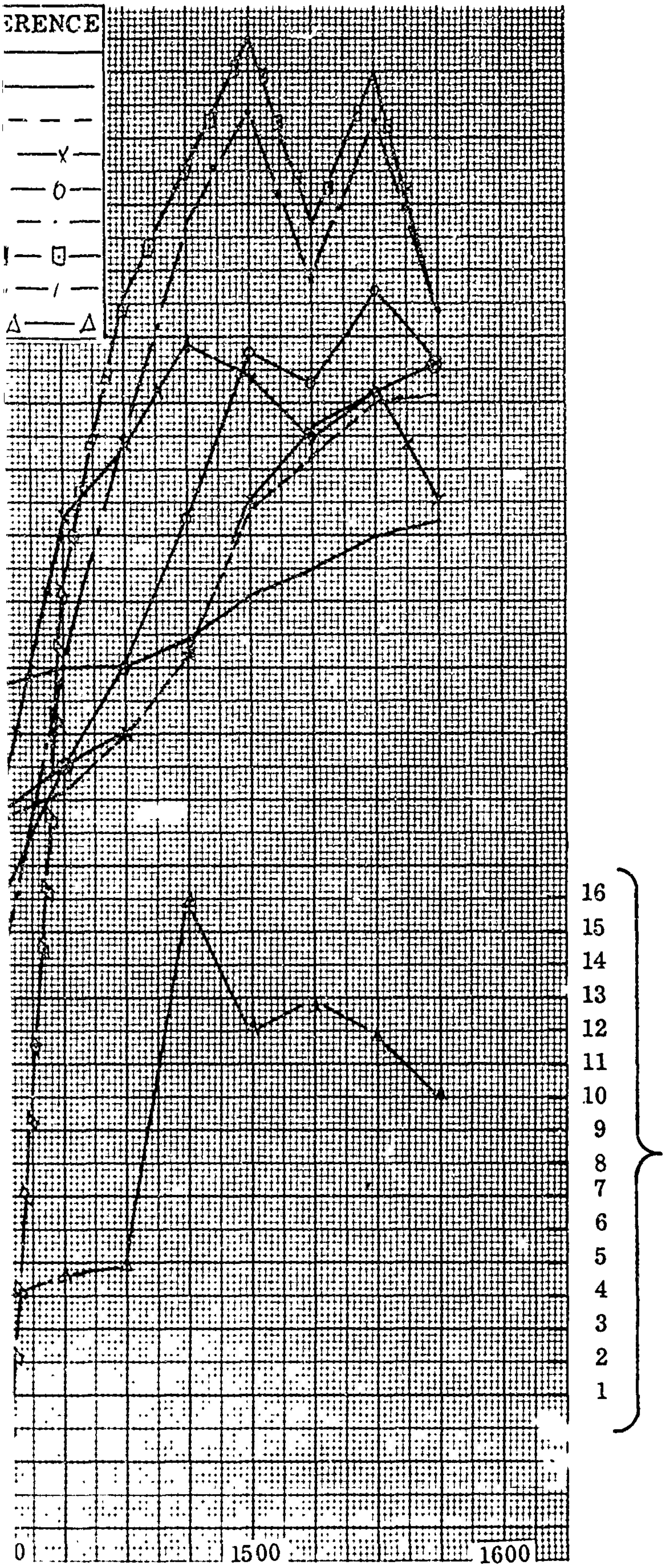
16
15
14
13
12
11
10
9
8
7
6
5
4
3
2
1

$\times 10^{-6}$ MM Hg

Figure 3-91. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test, Steps 1 and 2 (Sheet 2 of 3)

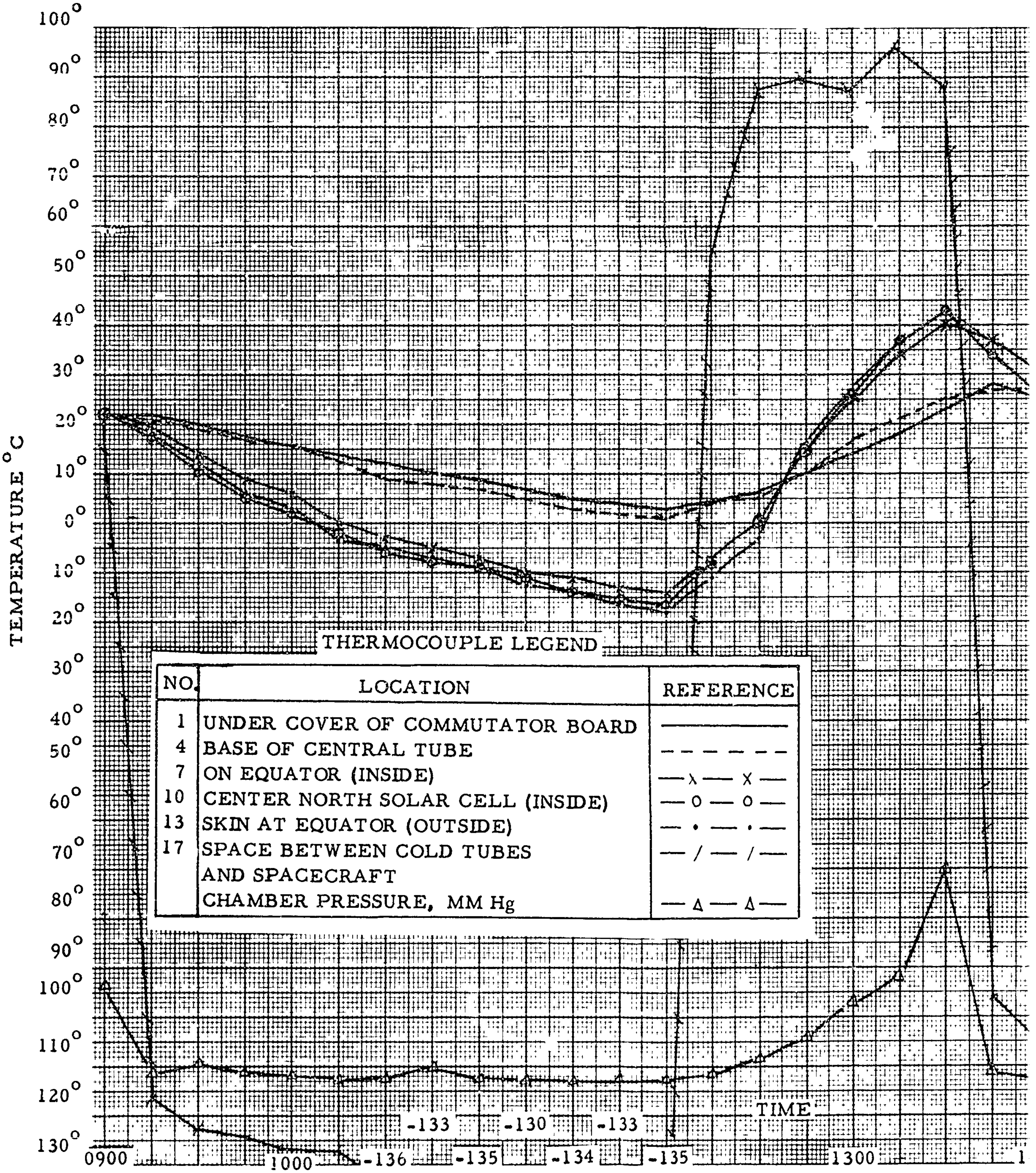


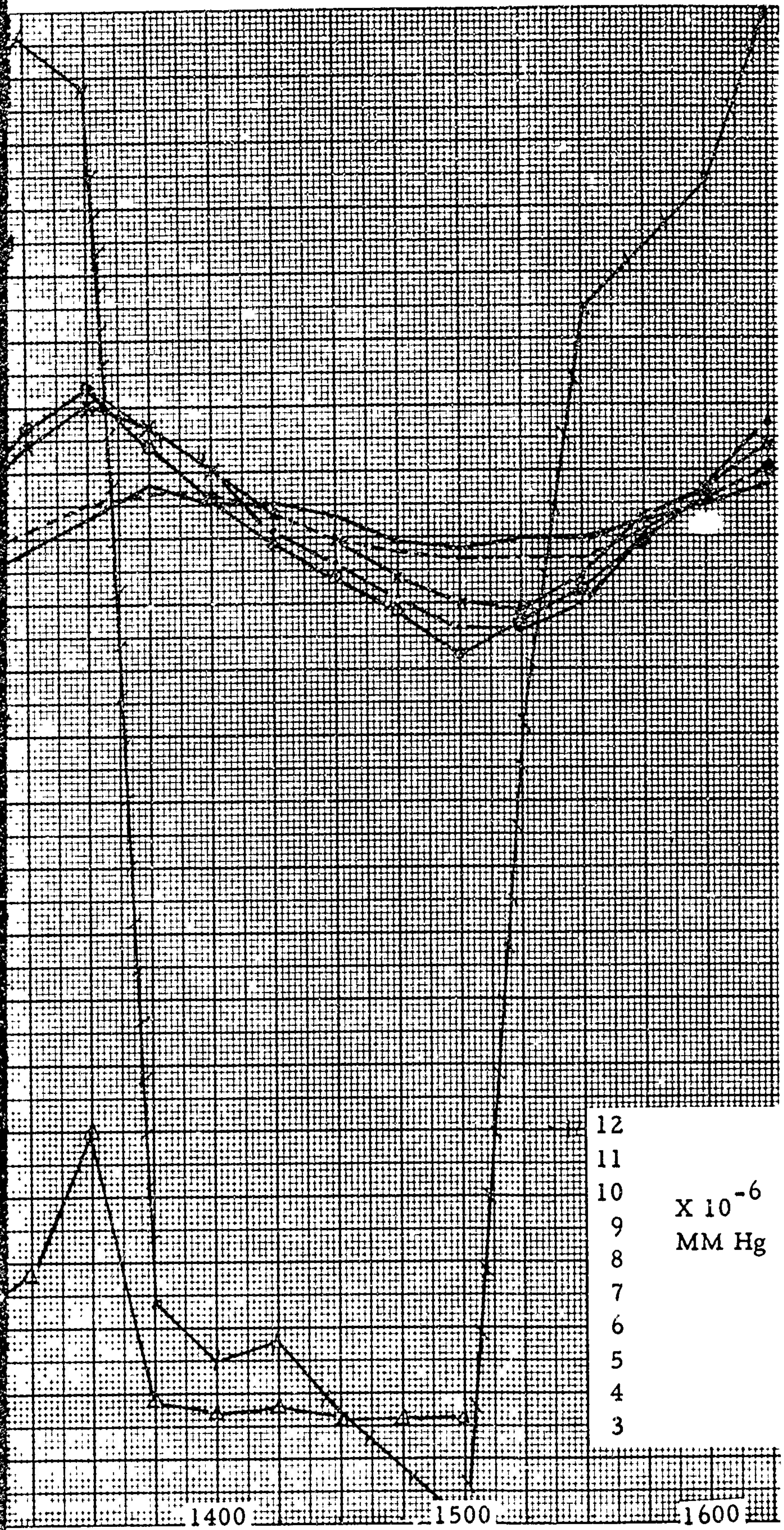
THERMOCOUPLE LEGEND
NO. | LOCATION | REFERENCE
3 CENTER OF RECEIVER CHASSIS
6 BETWEEN SOLAR CELLS (INSIDE)
9 NORTHERN REGION (INSIDE SKIN)
12 NORTH SOLAR CELL (OUTSIDE)
15 SOUTH SOLAR CELL
17 SPACE BETWEEN COLD TUBES AND SPACECRAFT
18 CHAMBER TEMPERATURE
CHAMBER TEMPERATURE MM, Hg
TEMPERATURE °C
+
-
110°
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
100°
110°
120°
1100
1200
1300
1400
1500
TIME



DATE	8/8/61
TIME	1100 TO 1545
RUN NO.	1
READINGS	20
REF. DATA BOOK	1312

Figure 3-91. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test, Steps 1 and 2 (Sheet 3 of 3)

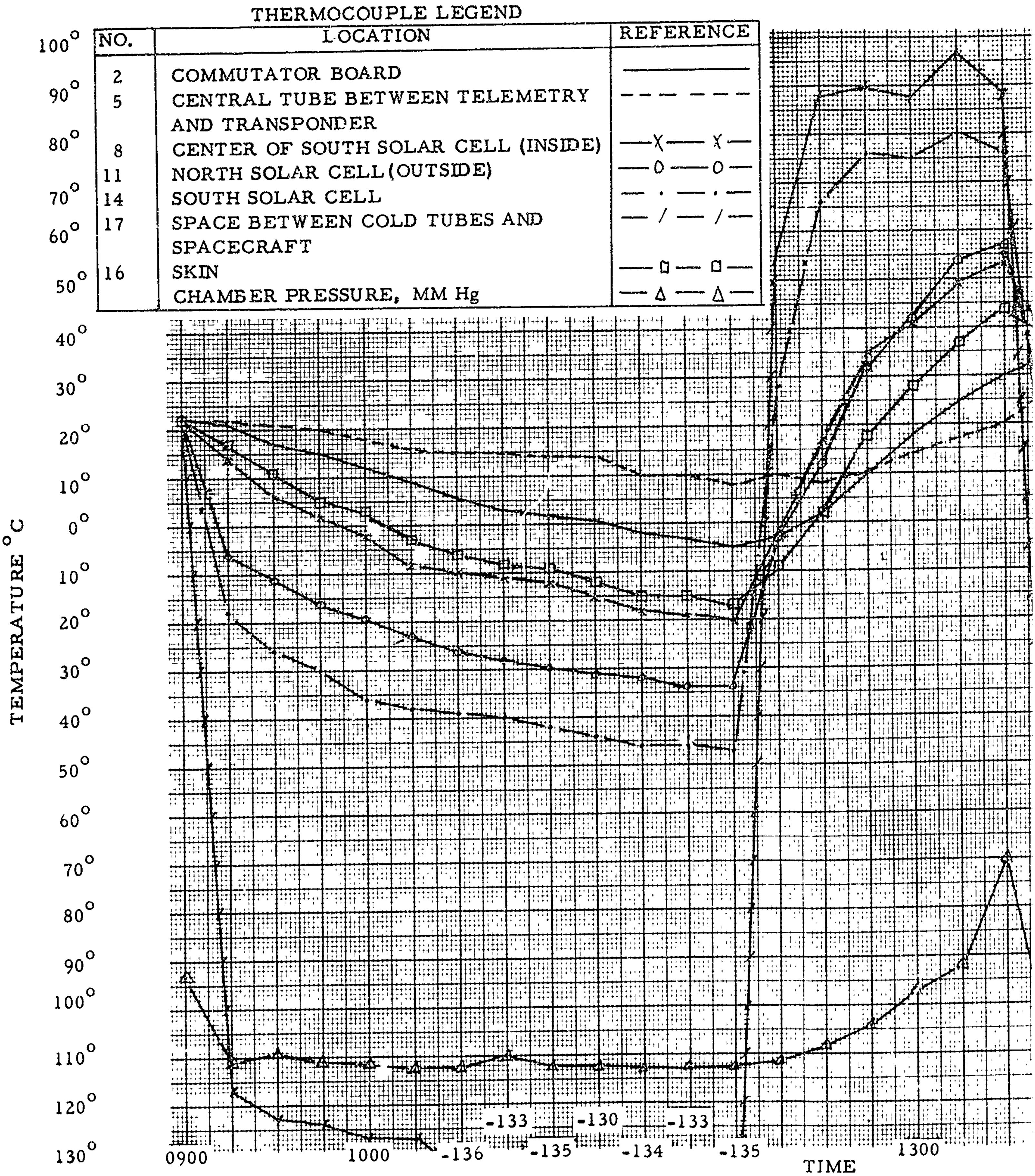

TEMPERATURE °C
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
100°
110°
120°
130°
THERMOCOUPLE LEGEND
NO. | LOCATION | REFERENCE
1 UNDER COVER OF COMMUTATOR BOARD
4 BASE OF CENTRAL TUBE
7 ON EQUATOR (INSIDE)
10 CENTER NORTH SOLAR CELL (INSIDE)
13 SKIN AT EQUATOR (OUTSIDE)
17 SPACE BETWEEN COLD TUBES AND SPACECRAFT
CHAMBER PRESSURE, MM Hg
-133
-130
-133
-136
-135
-134
-135
TIME
0900
1000
1300



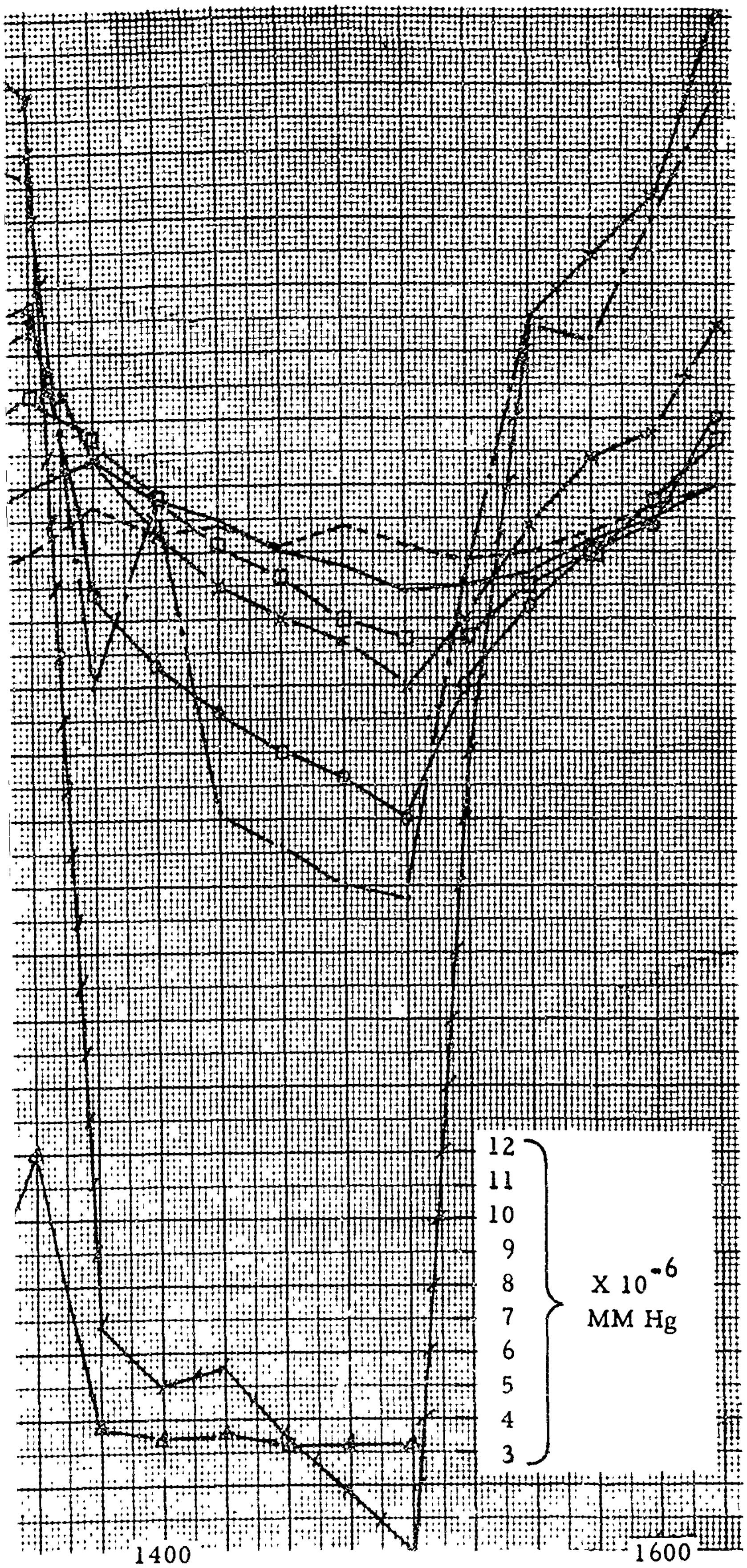
DATE
TIME

DATE	8/9/61
TIME	0900 to 1615
RUN NO.	2
READING	25
REF. DATA BOOK	1312

Figure 3-92. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test, Steps 3, 4, 5, and 6 (Sheet 1 of 3)

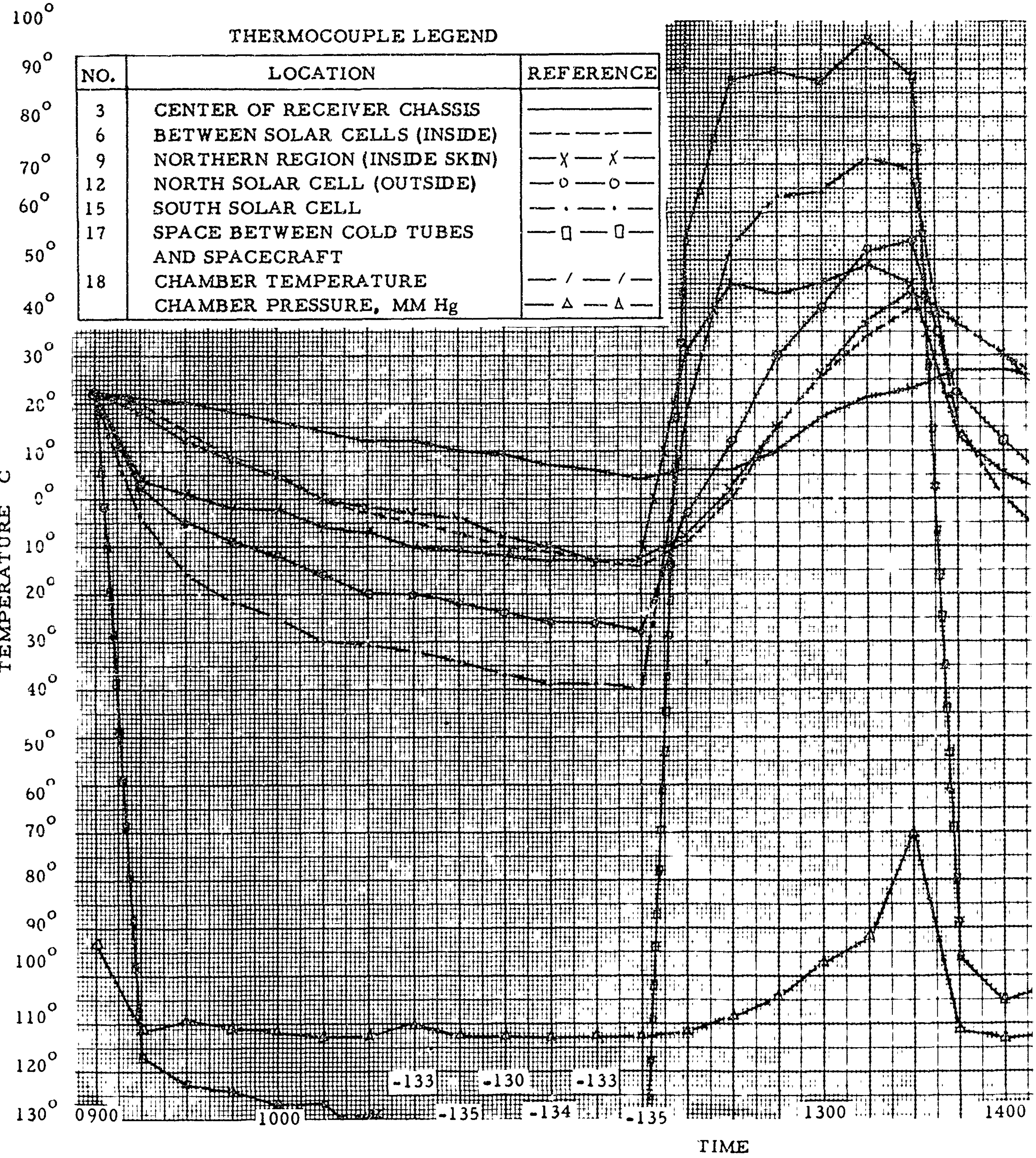

THERMOCOUPLE LEGEND
NO. | LOCATION | REFERENCE
2 COMMUTATOR BOARD
5 CENTRAL TUBE BETWEEN TELEMETRY AND TRANSPONDER
8 CENTER OF SOUTH SOLAR CELL (INSIDE)
11 NORTH SOLAR CELL (OUTSIDE)
14 SOUTH SOLAR CELL
17 SPACE BETWEEN COLD TUBES AND SPACECRAFT
16 SKIN
CHAMBER PRESSURE, MM Hg
TEMPERATURE °C
100° 90° 80° 70° 60° 50° 40° 30° 20° 10° 0° 10° 20° 30° 40° 50° 60° 70° 80° 90° 100° 110° 120° 130°
-133 -130 -133
0900 1000 -136 -135 -134 -135 1300
TIME

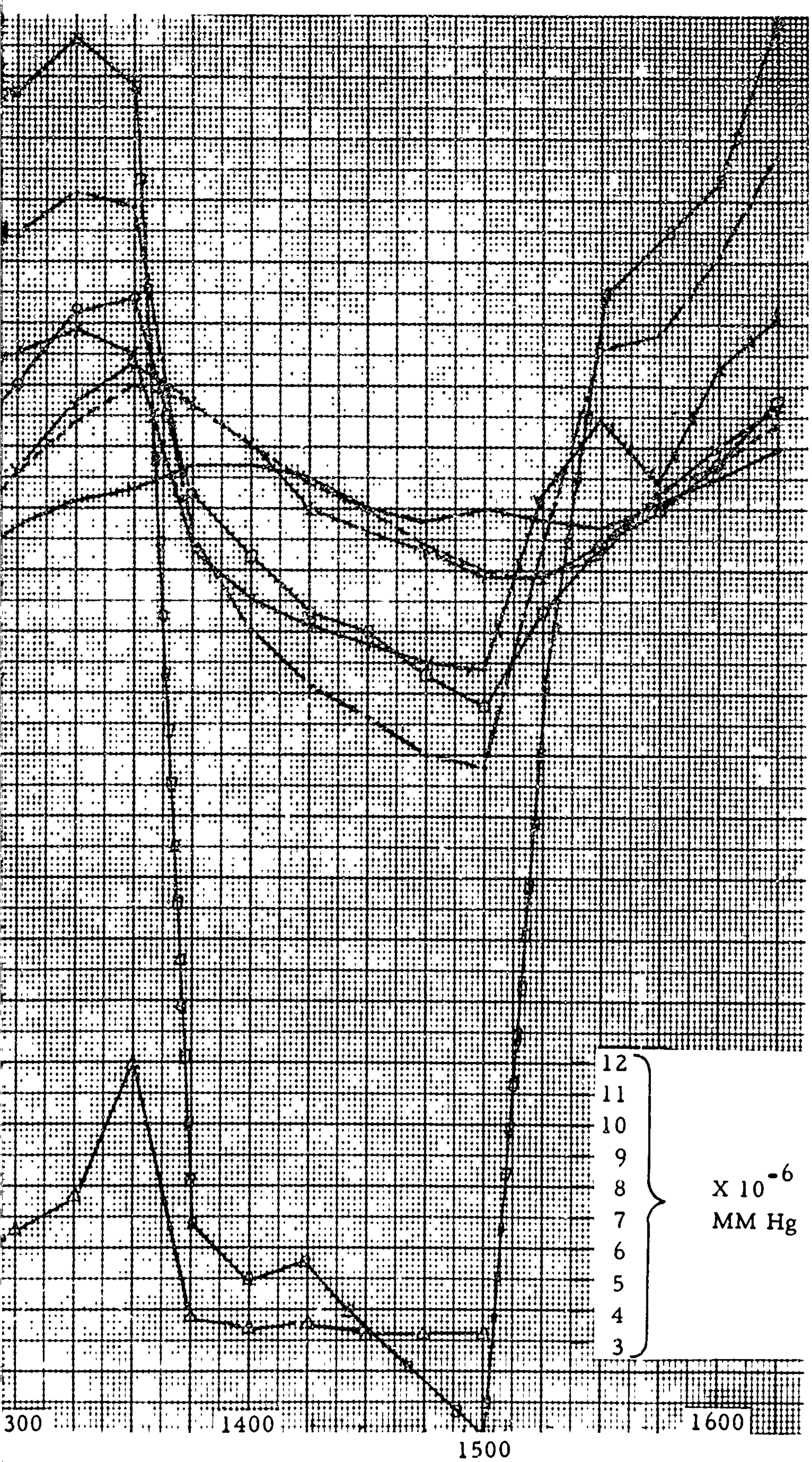
A



DATE	8/9/61
TIME	0900 to 1615
RUN NO.	2
READINGS	25
REF. DATA BOOK	1312

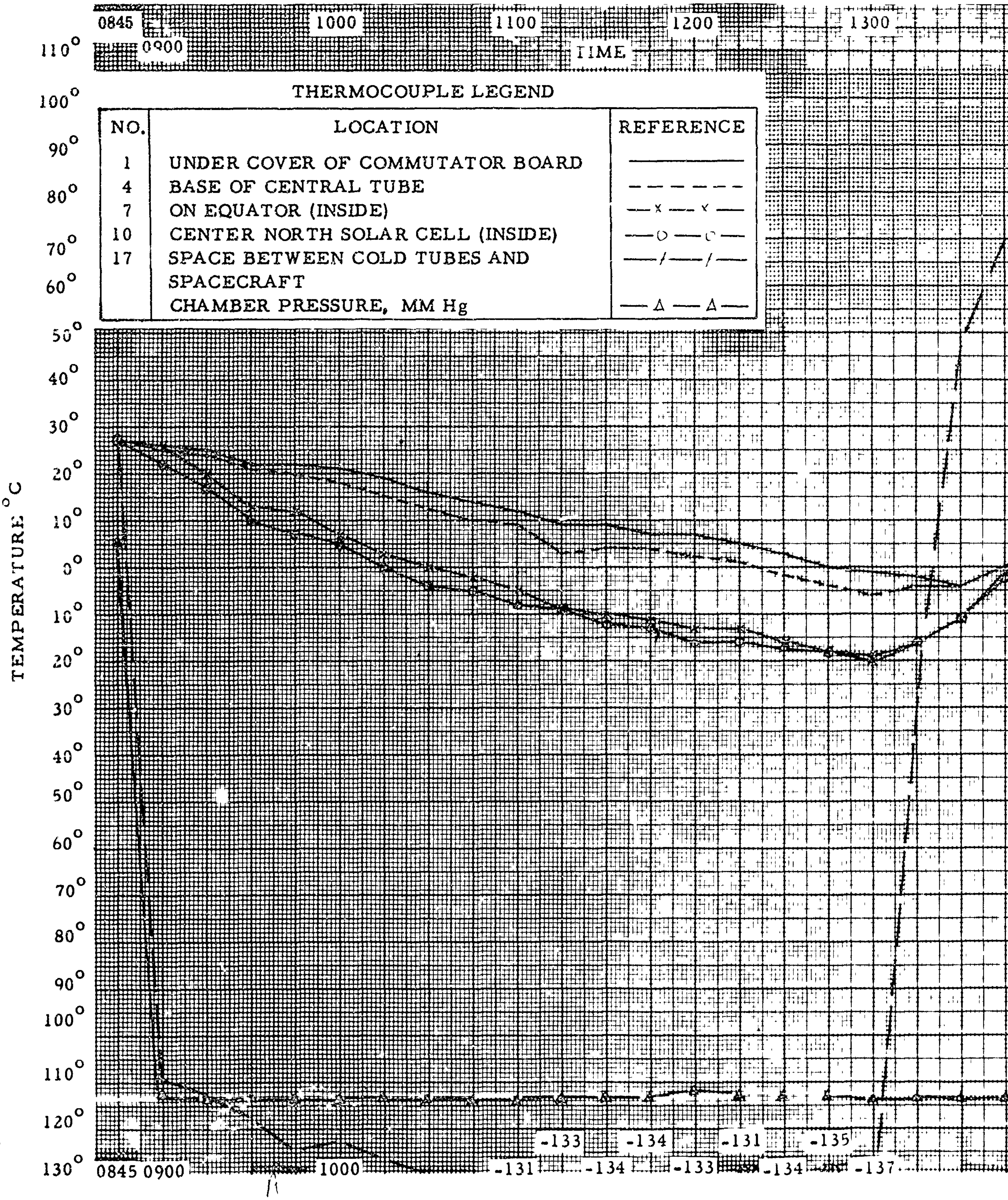
Figure 3-92. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test Steps 3, 4, 5, and 6 (Sheet 2 of 3)


THERMOCOUPLE LEGEND
NO. LOCATION REFERENCE
3 CENTER OF RECEIVER CHASSIS
6 BETWEEN SOLAR CELLS (INSIDE)
9 NORTHERN REGION (INSIDE SKIN)
12 NORTH SOLAR CELL (OUTSIDE)
15 SOUTH SOLAR CELL
17 SPACE BETWEEN COLD TUBES AND SPACECRAFT
18 CHAMBER TEMPERATURE
CHAMBER PRESSURE, MM Hg
TEMPERATURE °C
100° 90° 80° 70° 60° 50° 40° 30° 20° 10° 0° 10° 20° 30° 40° 50° 60° 70° 80° 90° 100° 110° 120° 130°
-133 -130 -133
-135 -134 -135
0900 1000 1300 1400
TIME

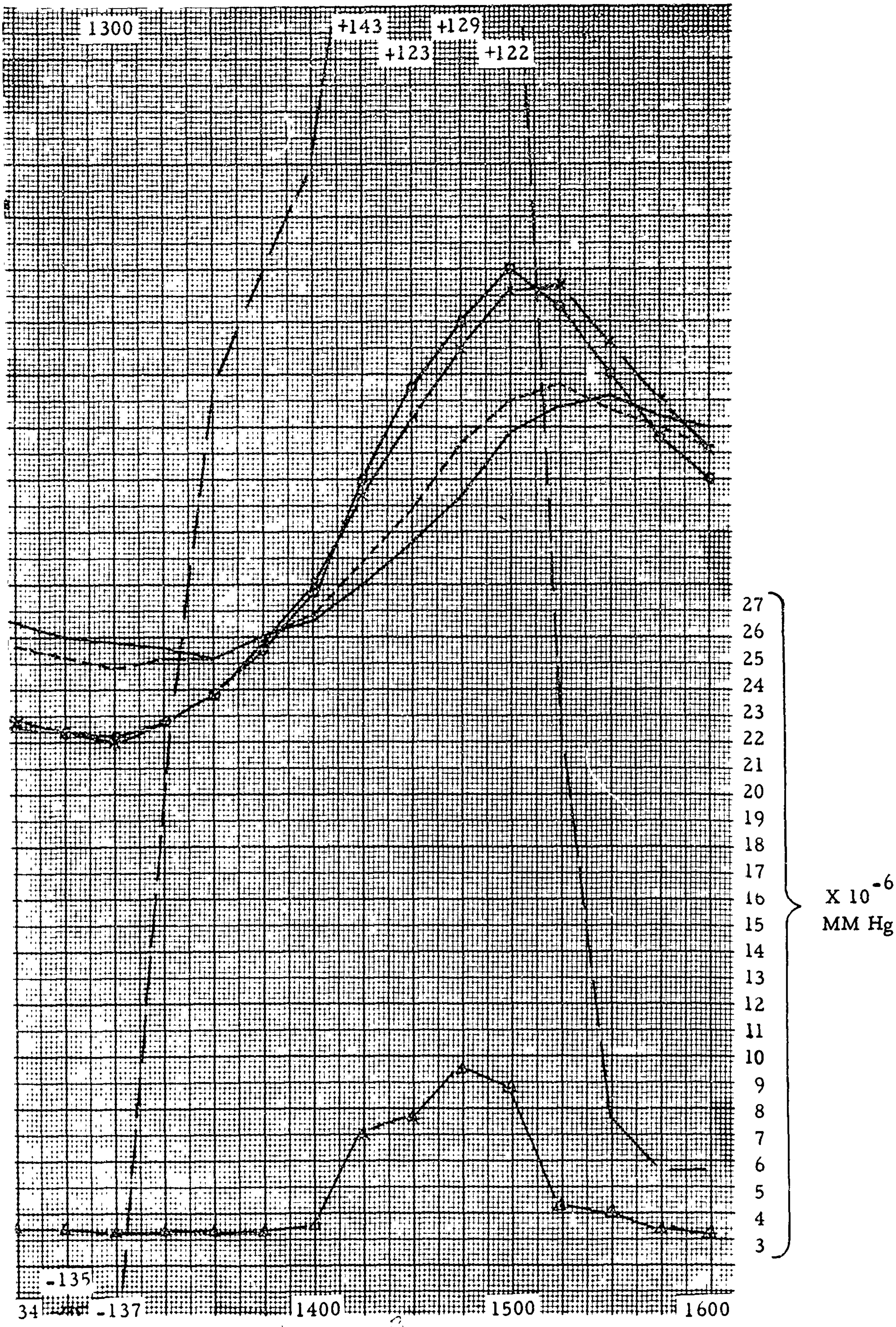


DATE 8/9/61
TIME 0900 to 1615
RUN NO. 2
READINGS 25
REF. DATA BOOK 1312

Figure 3-92. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test Steps 3, 4, 5, and 6 (Sheet 3 of 3)

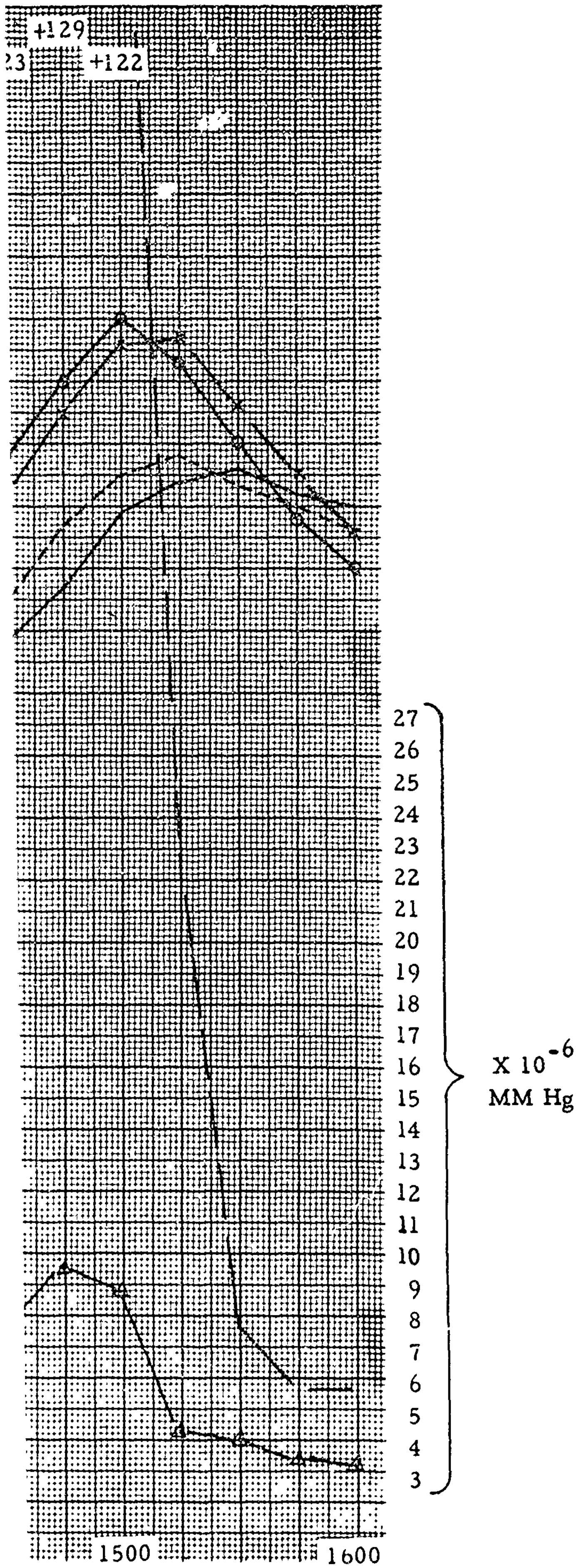


0845
0900
1000
1100
1200
1300
TIME
THERMOCOUPLE LEGEND
NO. | LOCATION | REFERENCE
1 UNDER COVER OF COMMUTATOR BOARD
4 BASE OF CENTRAL TUBE
7 ON EQUATOR (INSIDE)
10 CENTER NORTH SOLAR CELL (INSIDE)
17 SPACE BETWEEN COLD TUBES AND SPACECRAFT
CHAMBER PRESSURE, MM Hg
TEMPERATURE °C
110°
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
100°
110°
120°
130°
-133
-134
-131
-135
0845 0900
1000
-131
-134
-133
-134
-137



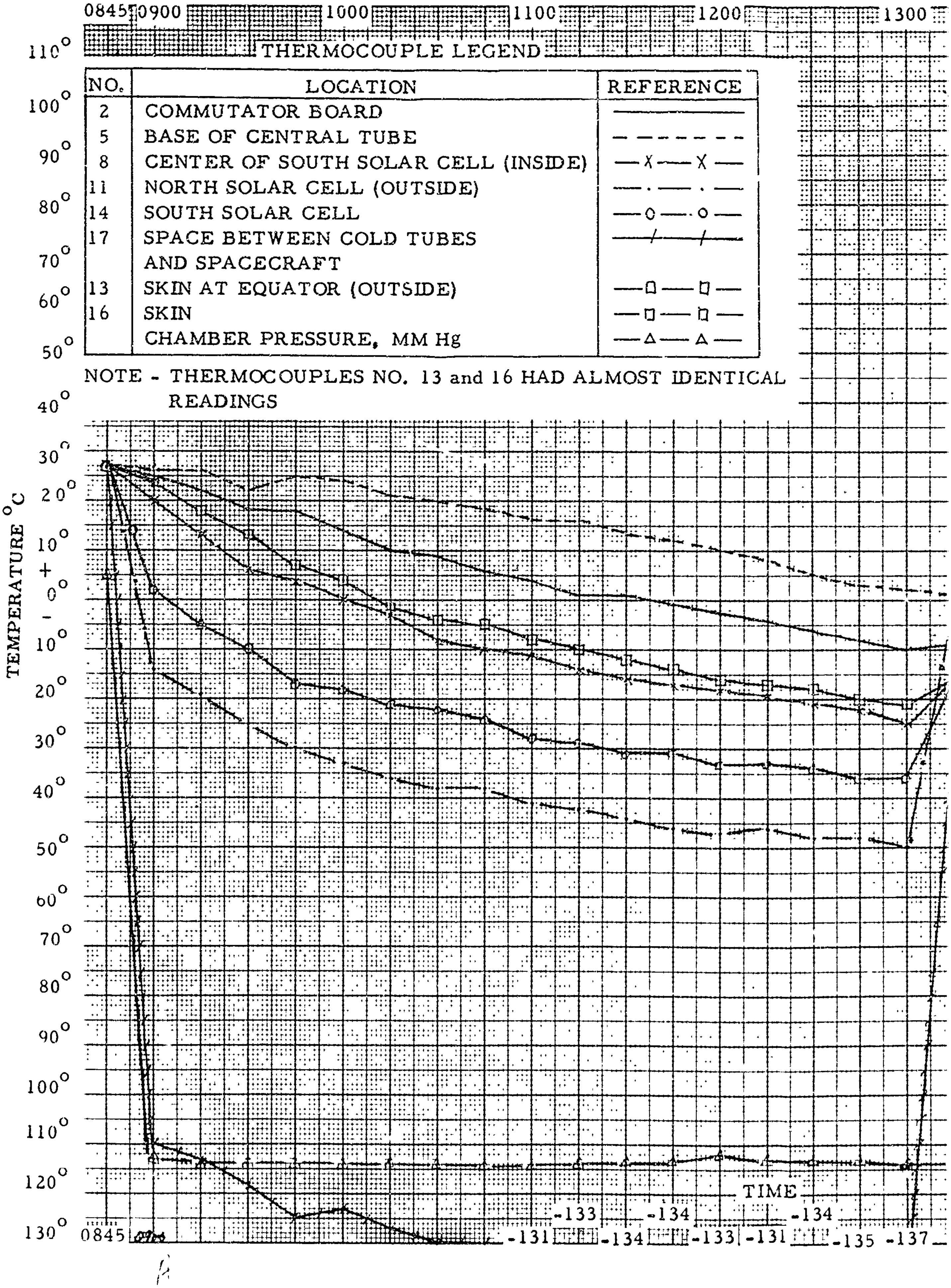
DATE
TIME
RUN NO.
READINGS
REF. DATA BOOK

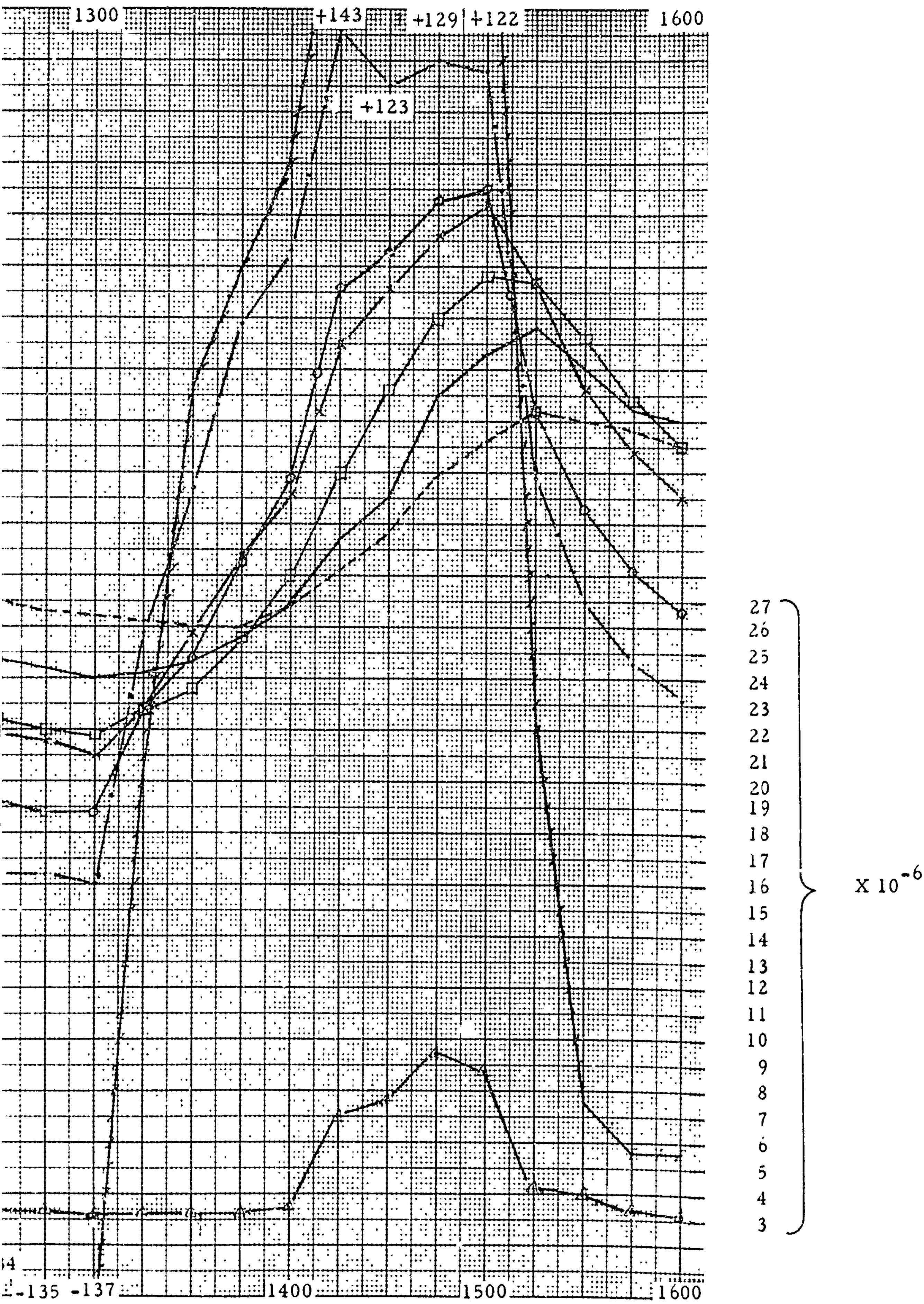
Figure 3-93. Tl
for Geodetic Spa
Thermal-Vacuur
and 9 (Sheet 1



DATE	8/10/61
TIME	0845 to 1600
RUN NO.	3
READINGS	29
REF. DATA BOOK	1312

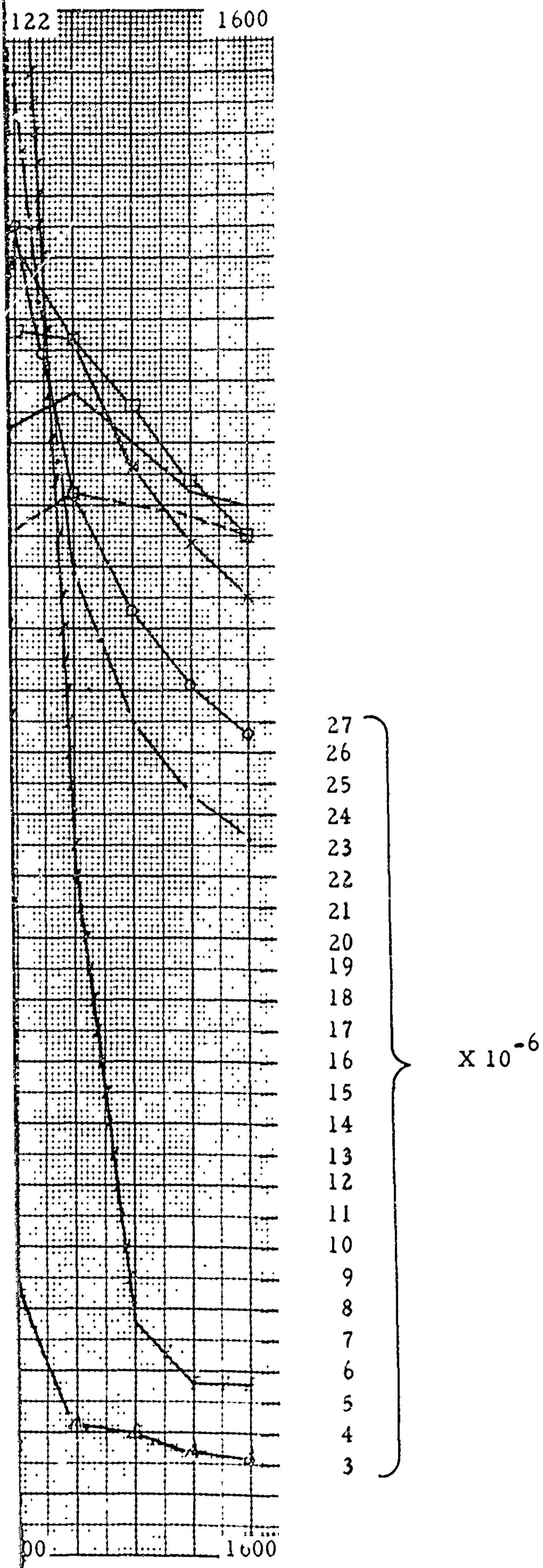
Figure 3-93. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test, Steps 7, 8, and 9 (Sheet 1 of 3)


0845 0900 1000 1100 1200 1300
THERMOCOUPLE LEGEND
NO. LOCATION REFERENCE
2 COMMUTATOR BOARD
5 BASE OF CENTRAL TUBE
8 CENTER OF SOUTH SOLAR CELL (INSIDE)
11 NORTH SOLAR CELL (OUTSIDE)
14 SOUTH SOLAR CELL
17 SPACE BETWEEN COLD TUBES AND SPACECRAFT
13 SKIN AT EQUATOR (OUTSIDE)
16 SKIN
CHAMBER PRESSURE, MM Hg
NOTE - THERMOCOUPLES NO. 13 and 16 HAD ALMOST IDENTICAL READINGS
TEMPERATURE °C
+
-
110° 100° 90° 80° 70° 60° 50° 40° 30° 20° 10° 0° 10° 20° 30° 40° 50° 60° 70° 80° 90° 100° 110° 120° 130°
TIME
-133 -134 -134
0845 -131 -134 -133 -131 -135 -137



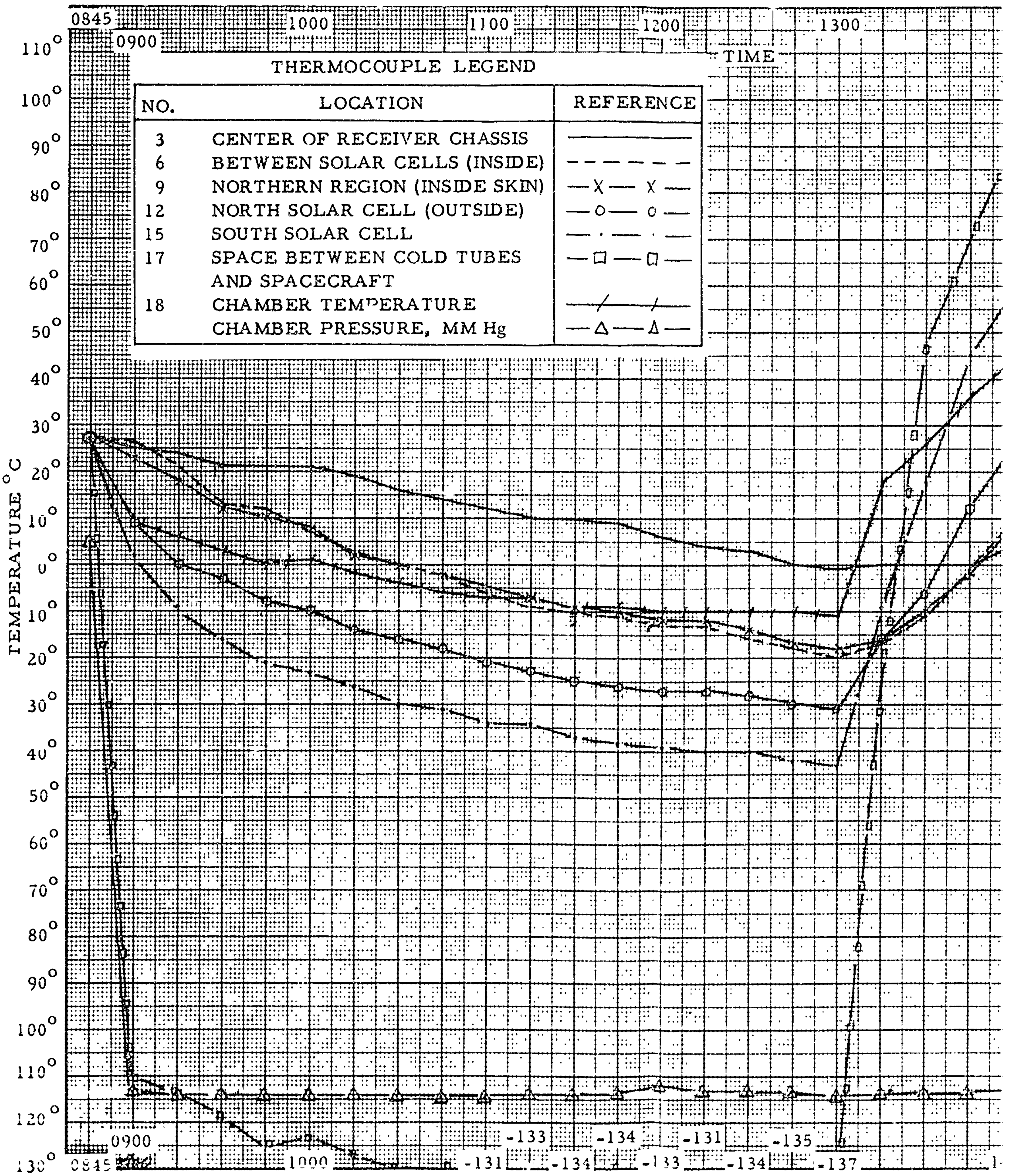
DATE
TIME
RUN NO
READIN
REF. D

Figur
for G
Ther
and 9

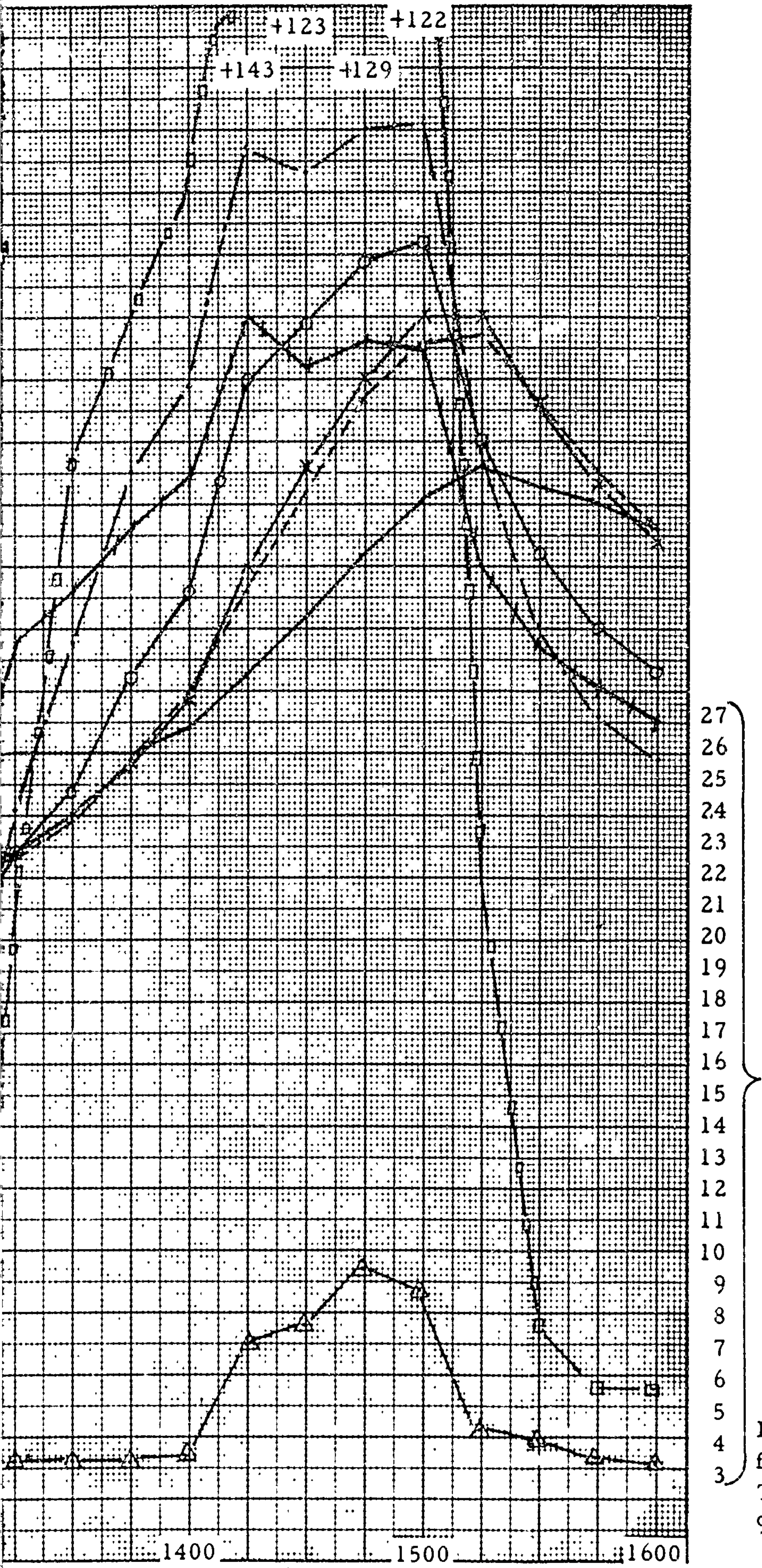


DATE	8/10/61
TIME	0845 to 1600
RUN NO.	3
READINGS	29
REF. DATA BOOK	1312

Figure 3-93. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test Steps 7, 8, and 9 (Sheet 2 of 3)

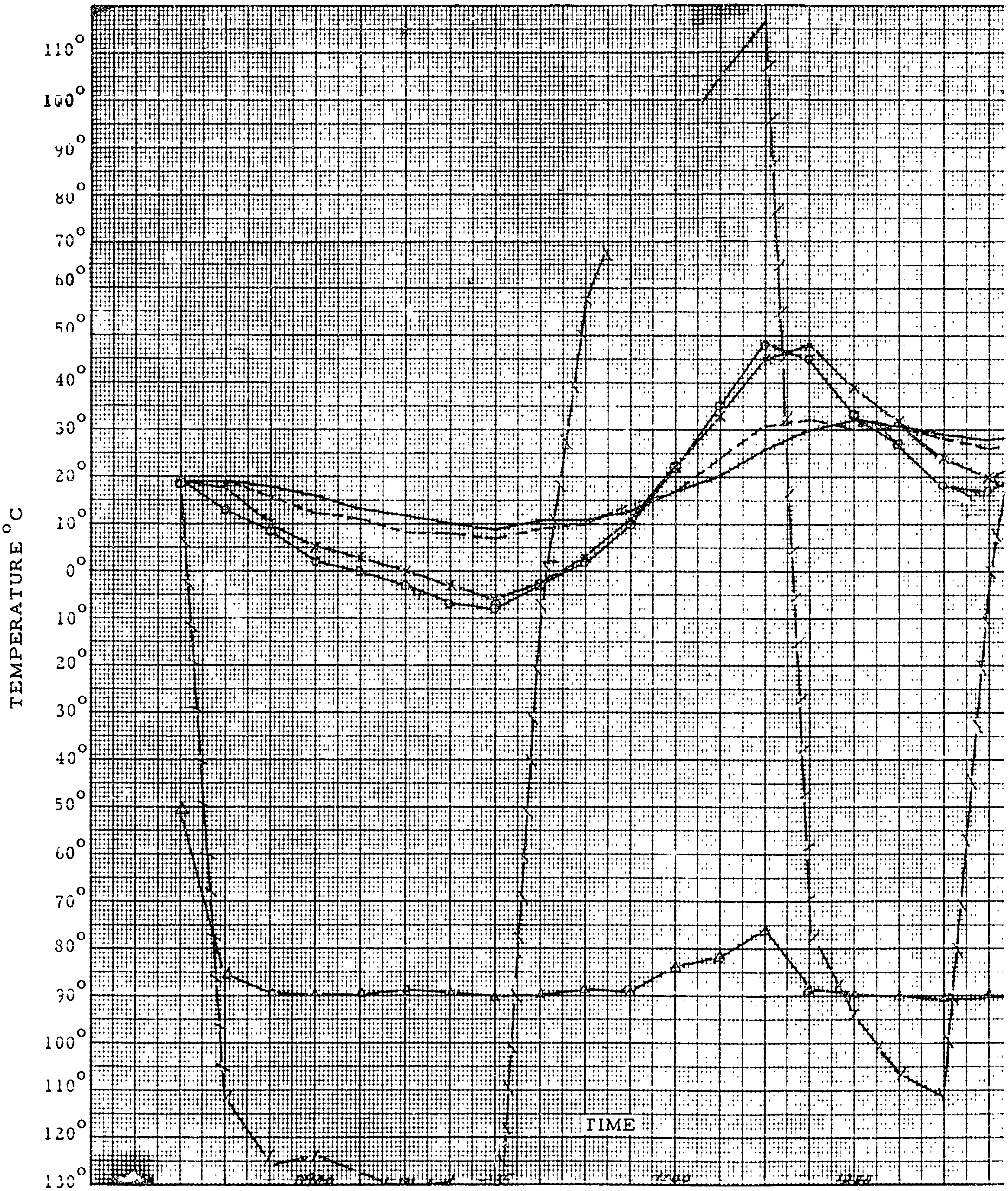


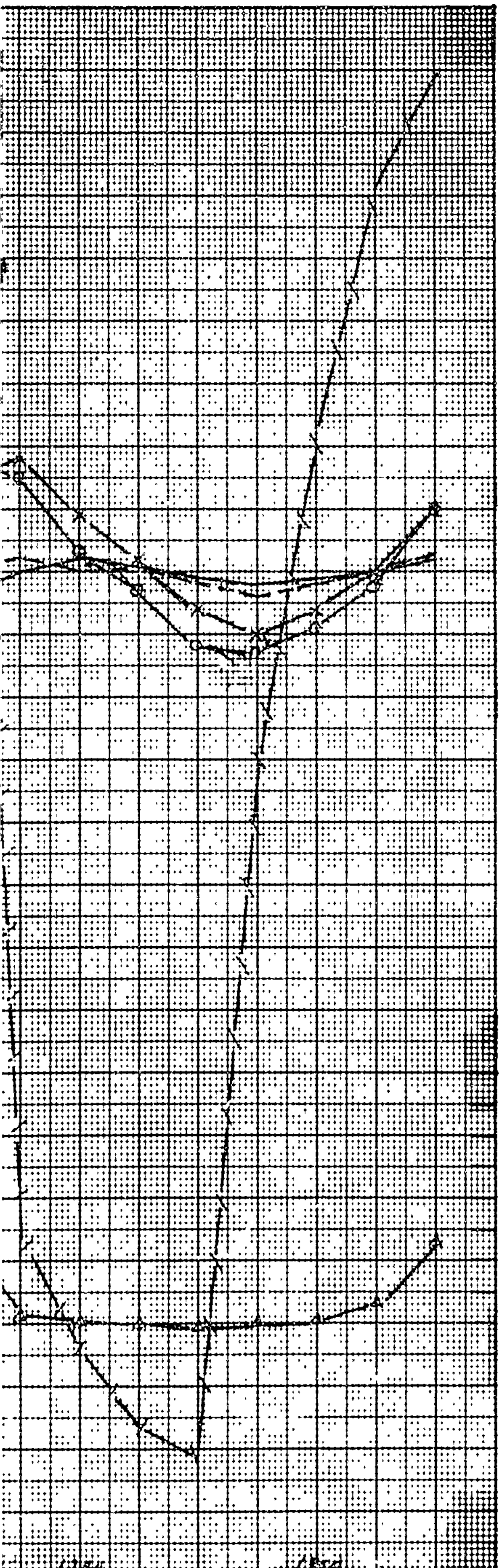
0845
0900
1000
1100
1200
1300
TIME
THERMOCOUPLE LEGEND
NO. | LOCATION | REFERENCE
3 CENTER OF RECEIVER CHASSIS
6 BETWEEN SOLAR CELLS (INSIDE)
9 NORTHERN REGION (INSIDE SKIN)
12 NORTH SOLAR CELL (OUTSIDE)
15 SOUTH SOLAR CELL
17 SPACE BETWEEN COLD TUBES AND SPACECRAFT
18 CHAMBER TEMPERATURE
CHAMBER PRESSURE, MM Hg
TEMPERATURE °C
110° 100° 90° 80° 70° 60° 50° 40° 30° 20° 10° 0° 10° 20° 30° 40° 50° 60° 70° 80° 90° 100° 110° 120° 130°
-133 -134 -131 -135
-131 -134 -133 -134 -137



DATE	8/10/61
TIME	0845 to 1600
RUN NO.	3
READINGS	29
REF. DATA BOOK	1312

Figure 3-93. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test Steps 7, 8 and 9 (Sheet 3 of 3)


TEMPERATURE °C
110°
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
100°
110°
120°
130°
TIME



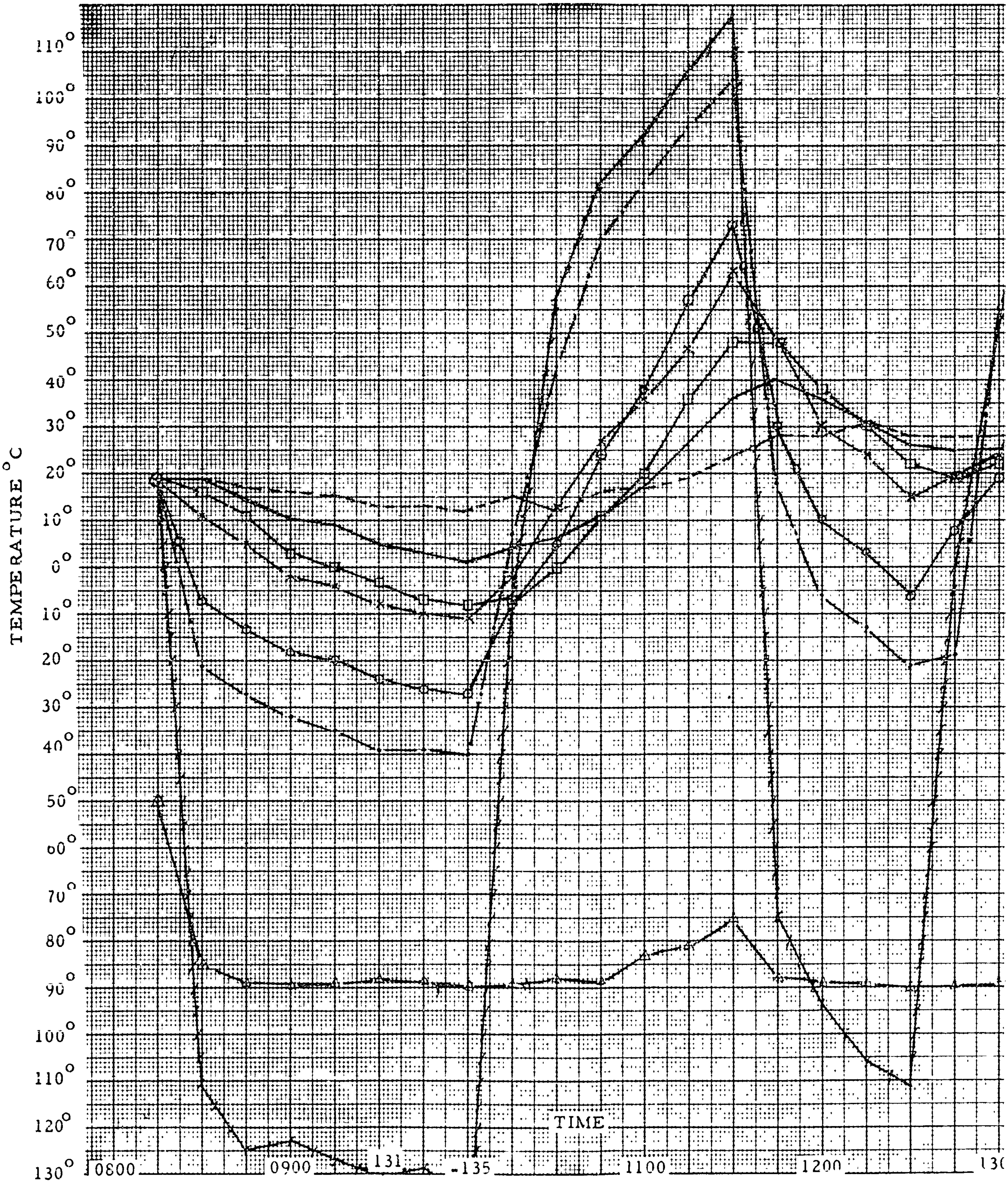
DATE	8/11/61
TIME	0815 to 1330
RUN NO.	4
READINGS	22
REF. DATA BOOK	1312

THERMOCOUPLE LEGEND

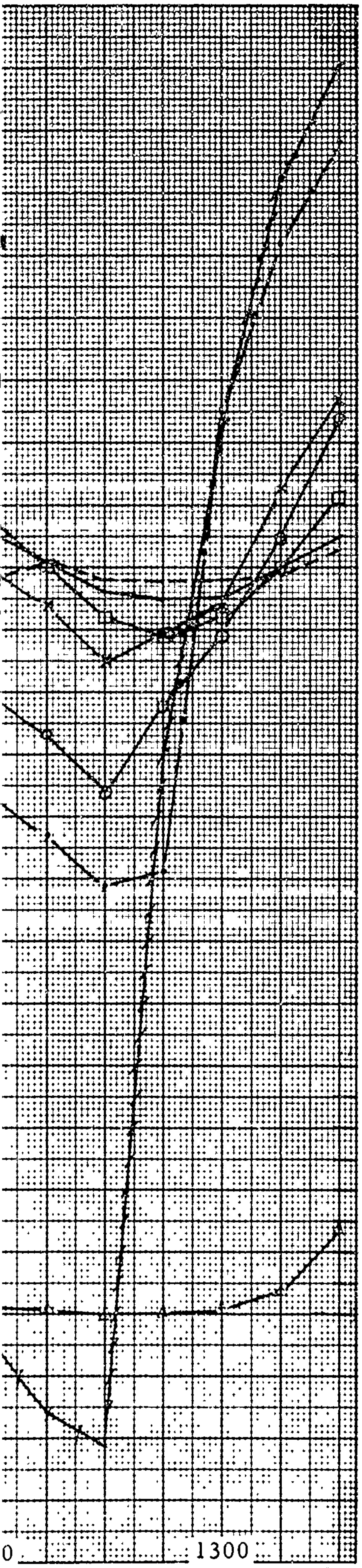
NO.	LOCATION	REFERENCE
1	UNDER COVER OF COMMUTATOR BOARD	————————
4	BASE OF CENTRAL TUBE	— — — — —
7	ON EQUATOR (INSIDE)	— x —— x —
10	CENTER NORTH SOLAR CELL (INSIDE)	— o —— o —
17	SPACE BETWEEN COLD TUBES AND SPACECRAFT	— / —— / —
	CHAMBER PRESSURE, MM HG	— Δ — Δ —

11
10
9
8 } X 10^{-6}
7 } MM HG
6
5
4
3

Figure 3-94. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test, Steps 10, 11, 12, and 13 (Sheet 1 of 3)



TEMPERATURE °C
110°
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
100°
110°
120°
130°
TIME
0800
0900
1100
1200
-135
131



DATE	8/11/61
TIME	0815 to 1330
RUN NO.	4
READINGS	22
REF. DATA BOOK	1312

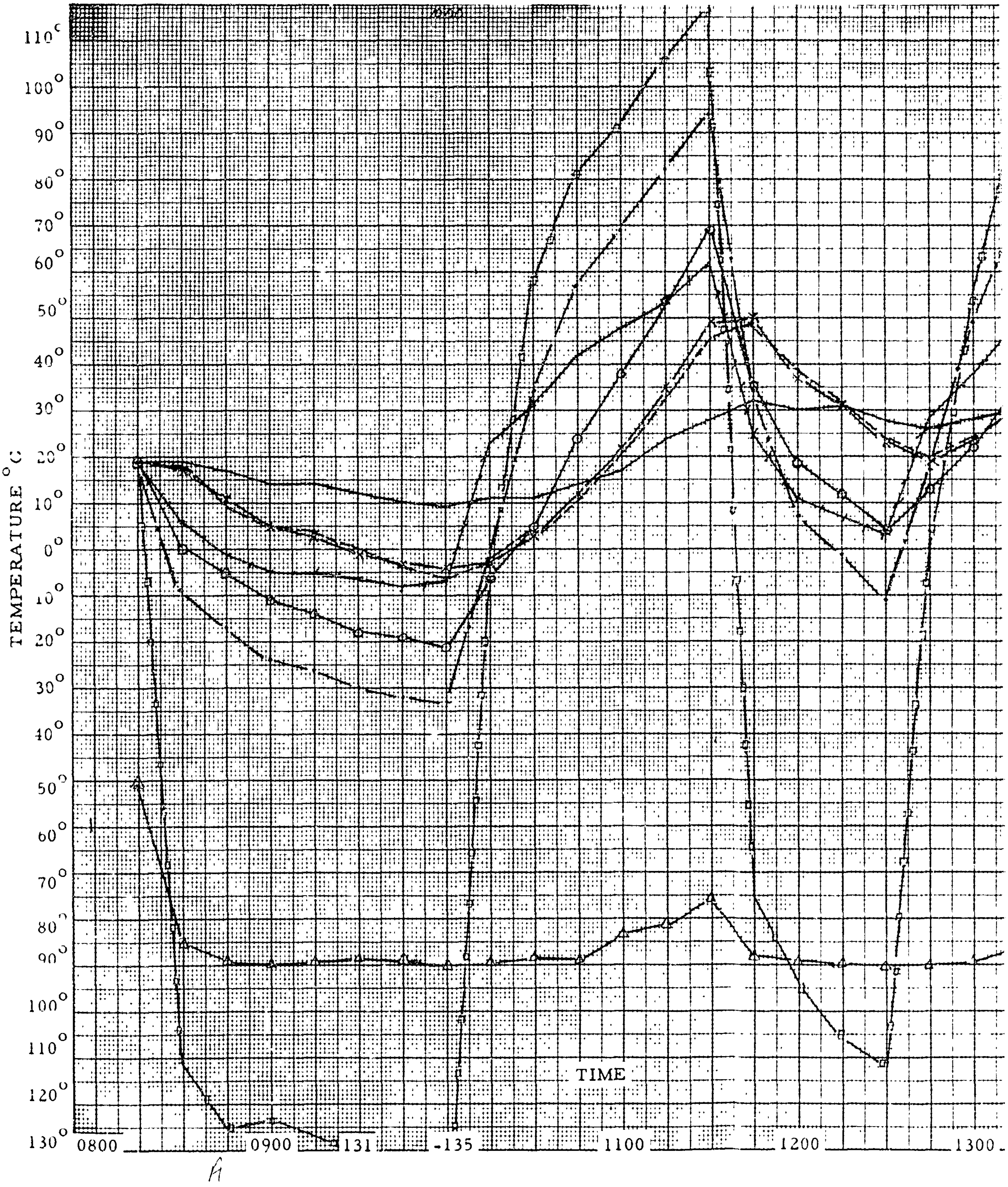
THERMOCOUPLE LEGEND

NO.	LOCATION	REFERENCE
2	COMMUTATOR BOARD	————————
5	CENTRAL TUBE BETWEEN TELEMETER AND TRANSPONDER	— — — — — —
8	CENTER OF SOUTH SOLAR CELL (INSIDE)	x —— x —— x
11	NORTH SOLAR CELL (OUTSIDE)	o —— o —— o
14	SOUTH SOLAR CELL	· —— · —— ·
17	SPACE BETWEEN COLD TUBES AND SPACECRAFT	+———+———+
13	SKIN AT EQUATOR	□ —— □ —— □
16	SKIN	□——□——□
	CHAMBER PRESSURE, MM HG	Δ —— Δ —— Δ

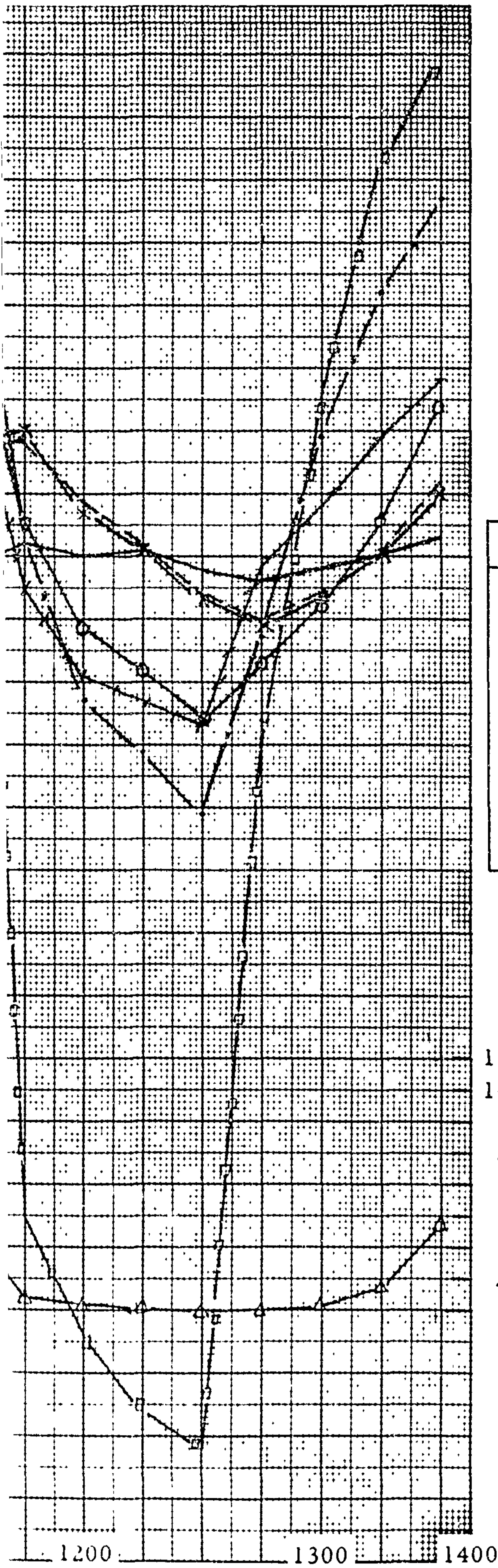
NOTE - THERMOCOUPLES NO. 13 and 16 HAD ALMOST IDENTICAL READINGS

11
10
9
8
7 $\Big\}$ X 10^{-6}
6 MM HG
5
4
3

Figure 3-94. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Tests Steps 10, 11, 12, and 13 (Sheet 2 of 3)



110°
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
100°
110°
120°
130°
TEMPERATURE °C
TIME
0800
0900
131
-135
1100
1200
1300



DATE	8/11/61
TIME	0815 to 1330
RUN NO.	4
READINGS	22
REF. DATA BOOK	1312

THERMOCOUPLE LEGEND

NO.	LOCATION	REFERENCE
3	CENTER OF RECEIVER CHASSIS	———————
6	BETWEEN SOLAR CELLS (INSIDE)	- - - - - - -
9	NORTHERN REGION (INSIDE SKIN)	— X — X —
12	NORTH SOLAR CELL (OUTSIDE)	— o — o —
15	SOUTH SOLAR CELL	— · — · —
17	SPACE BETWEEN COLD TUBES AND SPACECRAFT	— □ — □ —
18	CHAMBER TEMPERATURE	—/—/—
	CHAMBER PRESSURE	— Δ — Δ —

Figure 3-94. Thermocouple Responses for Geodetic Spacecraft No. 2 Thermal-Vacuum Test Steps 10, 11, 12, and 13 (Sheet 3 of 3)

apply vibration along the longitudinal axis. The frequency of vibration was then scanned from 5 to 50 cps at 0.95 inch D. A. or ±1.5 g (whichever limited in 1.66 minutes); 50 to 500 cps at ±7.3g in 1.66 minutes; 500 to 2000 cps at ±14g in one minute; and from 2000 to 3000 cps at ±36g or to the limits of the vibration machine in 0.5 minute. The frequency of vibration was swept at an approximate logarithmic rate.

(2) With the test specimen still mounted in the longitudinal axis, the vibration exciter response was equalized to within ±2 db of a flat power spectral density curve within the frequency band of 20 to 2000 cps. The equalization servo output voltage graph was plotted using a ±1.5g constant input. The acceleration clipping was adjusted so that 99.7 per cent of the true Gaussian distribution was realized and displacement clipping was adjusted to 0.5 inch D. A. The test specimen was then subjected to random vibration with a frequency band of 20 to 2000 cps at a 11.8g (rms) level for 8 minutes.

(3) The test specimen and fixture was then mounted to the vibration exciter in a position to apply vibration along one transverse axis. The frequency of vibration was then scanned from 5 to 50 cps at ±0.6g in 1.66 minutes; 50 to 500 cps at ±1.4g in 1.66 minutes; 500 to 2000 cps ±2.8g in one minute; and from 2000 to 3000 cps at ±11.3g or to the limits of the vibration machine in 0.6 minute. The frequency of vibration was swept at an approximate logarithmic rate.

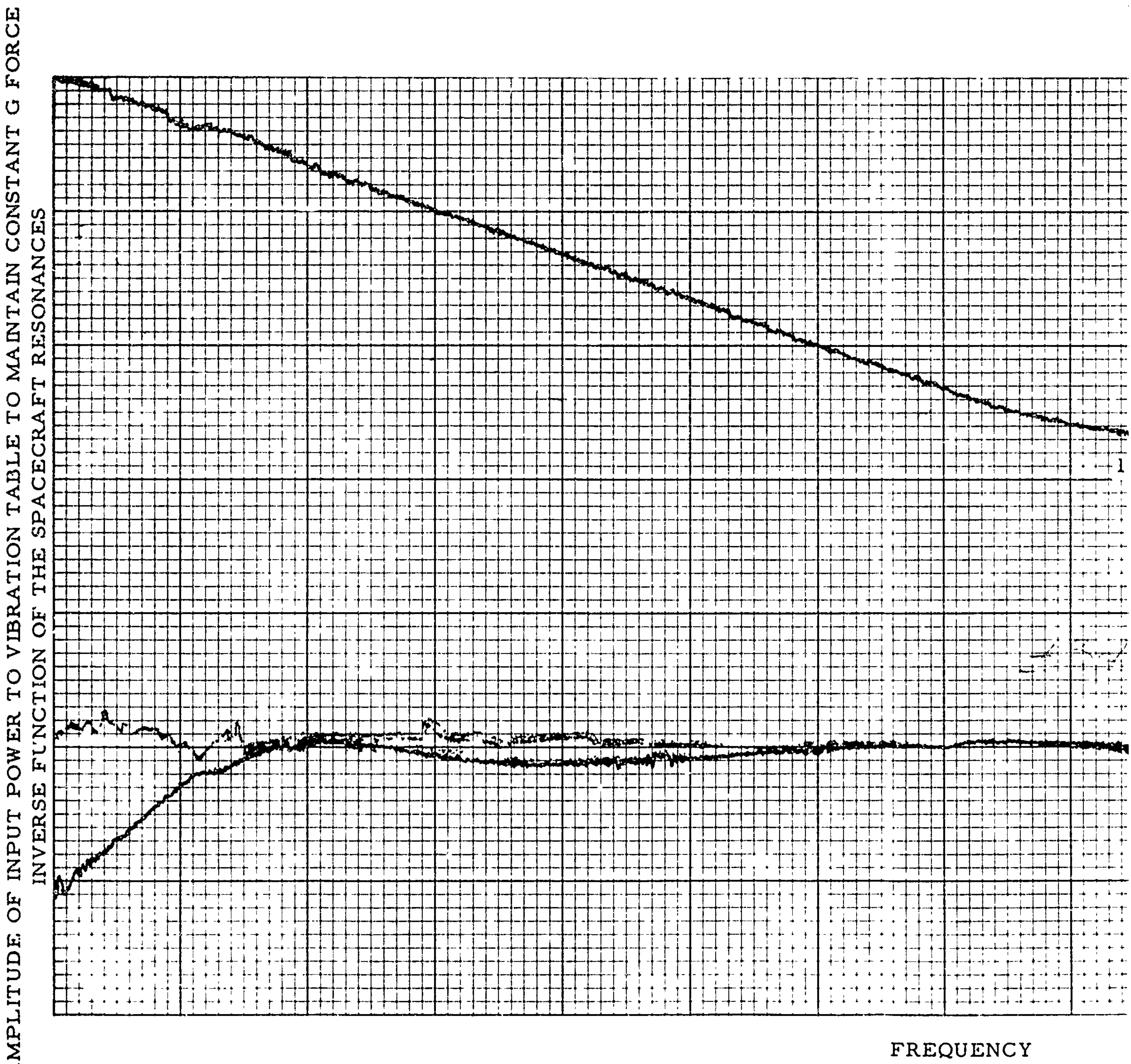
(4) With the test specimen still mounted in the transverse axis the vibration exciter response was equalized to within ±1.5 db of a flat power spectral density curve within the frequency band of 100 to 2000 cps. The acceleration clipping was adjusted so that 99.7 per cent of the true Gaussian distribution was realized and displacement clipping was adjusted to 0.5 inch D. A. The equalization servo output voltage graph was plotted using a ±0.6g constant input. The test specimen was then subjected to random vibration with a frequency band of 100 to 2000 cps at a 11.6g (rms) level for 8 minutes. The vibration test was completed in accordance with the test procedure. During and at the conclusion of the vibration test, the test specimen was examined for evidence of damage. As mechanical damage was not apparent, it was concluded that the Spacecraft had successfully completed the vibration testing. See figures 3-95 and 3-96 for a plot of the resonances.

3.13.4.3.2 <u>Acceleration Tests.</u> Acceleration tests were performed at General Dynamics, San Diego Division, San Diego, California on 21 July 1961. The Spacecraft successfully endured 35 g acceleration with overshoot to 37.8 g in the thrust axis with 1 g in the transverse axis for a one-minute period. Acceleration was applied for several minutes previous to the one-minute period to bring the centrifuge up to speed. Figures 3-83 and 3-84 give details of the test-mounting in the centrifuge.

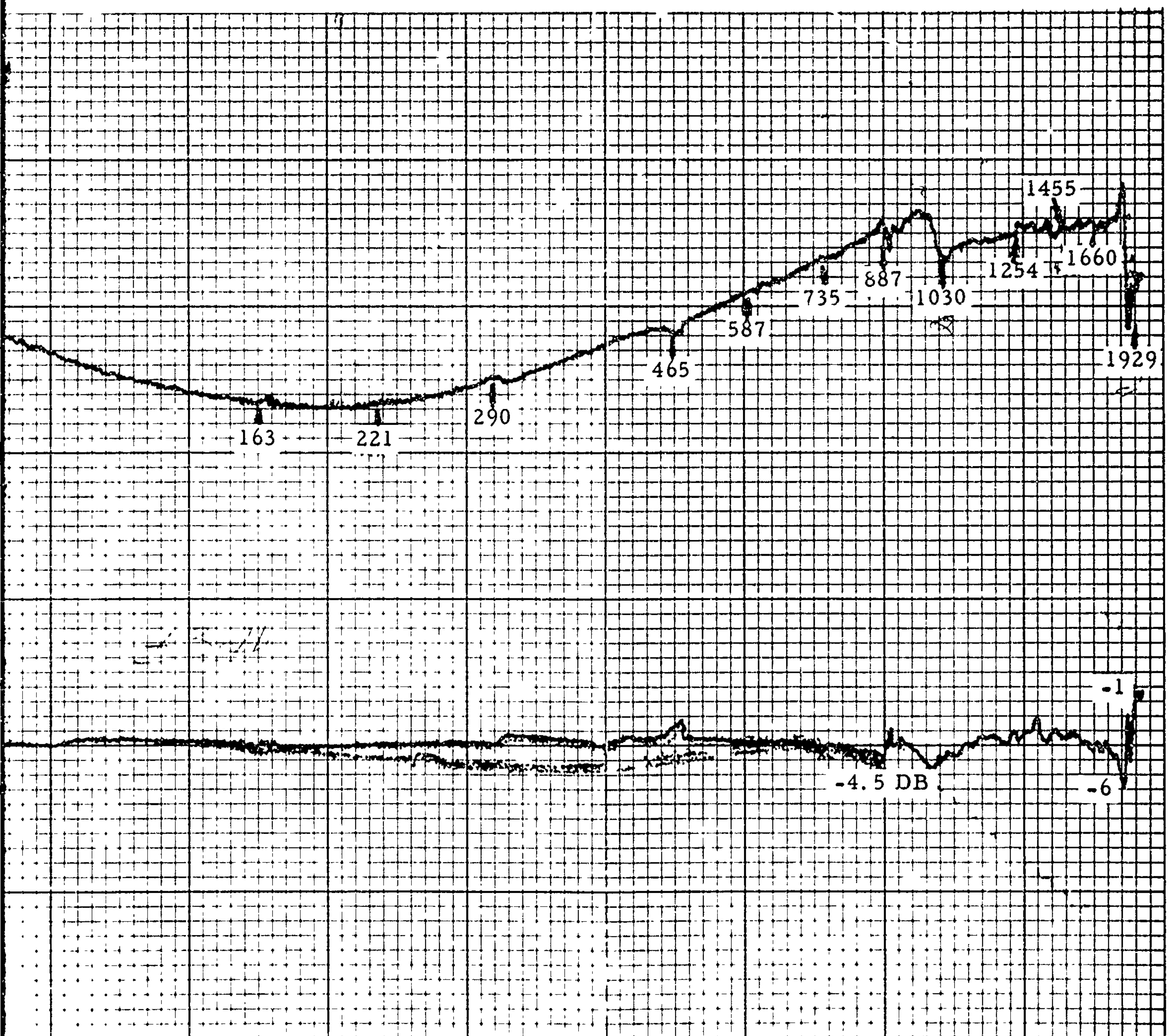
3.13.4.3.3 <u>α/ϵ Tests.</u> A test of the α/ϵ for the Spacecraft was performed at the Applied Physics Laboratory of Johns Hopkins University in Silver Springs, Maryland on 18 August 1961. The objective of the α/ϵ tests was to determine the ratio of absorptivity to emissivity actually achieved on the spacecraft outer shell. To accomplish this, the Spacecraft was referenced to a sphere that was painted flat black so as to have an α/ϵ of 1.0. The α/ϵ ratio found for the satellite using six thermocouple locations on the inside of the skin ranged from 0.87 to 1.19 with an average for the six locations of 1.00. This value compares favorably with the expected calculated value of 1.03. Applying a correction factor for the chamber and using 1.5 as the emissivity of the Spacecraft reduces the average α/ϵ to 0.97.

3.13.4.3.3.1 The importance of the α/ϵ ratio lies in its use to control the thermal balance of the satellite, thereby controlling its temperature. Thus, if it were feasible to simulate this radiation balance under actual vacuum conditions, the temperature achieved by the satellite shell could be compared to the temperature achieved by a sphere whose α/ϵ was known. The equations governing these relationships and the correction factors for the vacuum chamber walls, etc., are developed as follows.

3.13.4.3.3.2 For a body totally enclosed in an enclosure whose surface area (A_w) is large compared to the surface area of the body (A_{REF}), the energy balance at equilibrium may be written as:

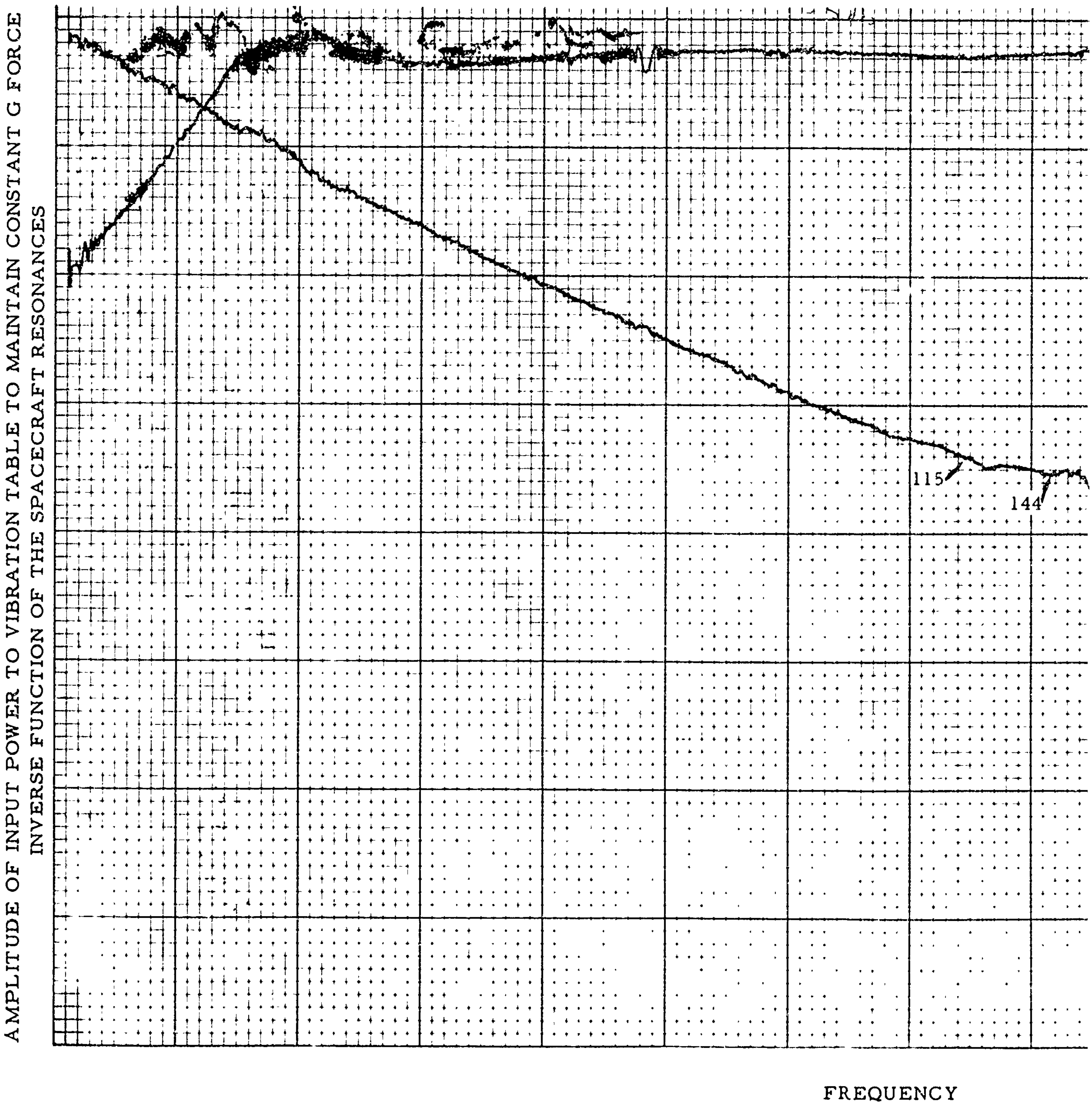

AMPLITUDE OF INPUT POWER TO VIBRATION TABLE TO MAINTAIN CONSTANT G FORCE
INVERSE FUNCTION OF THE SPACECRAFT RESONANCES
1
FREQUENCY

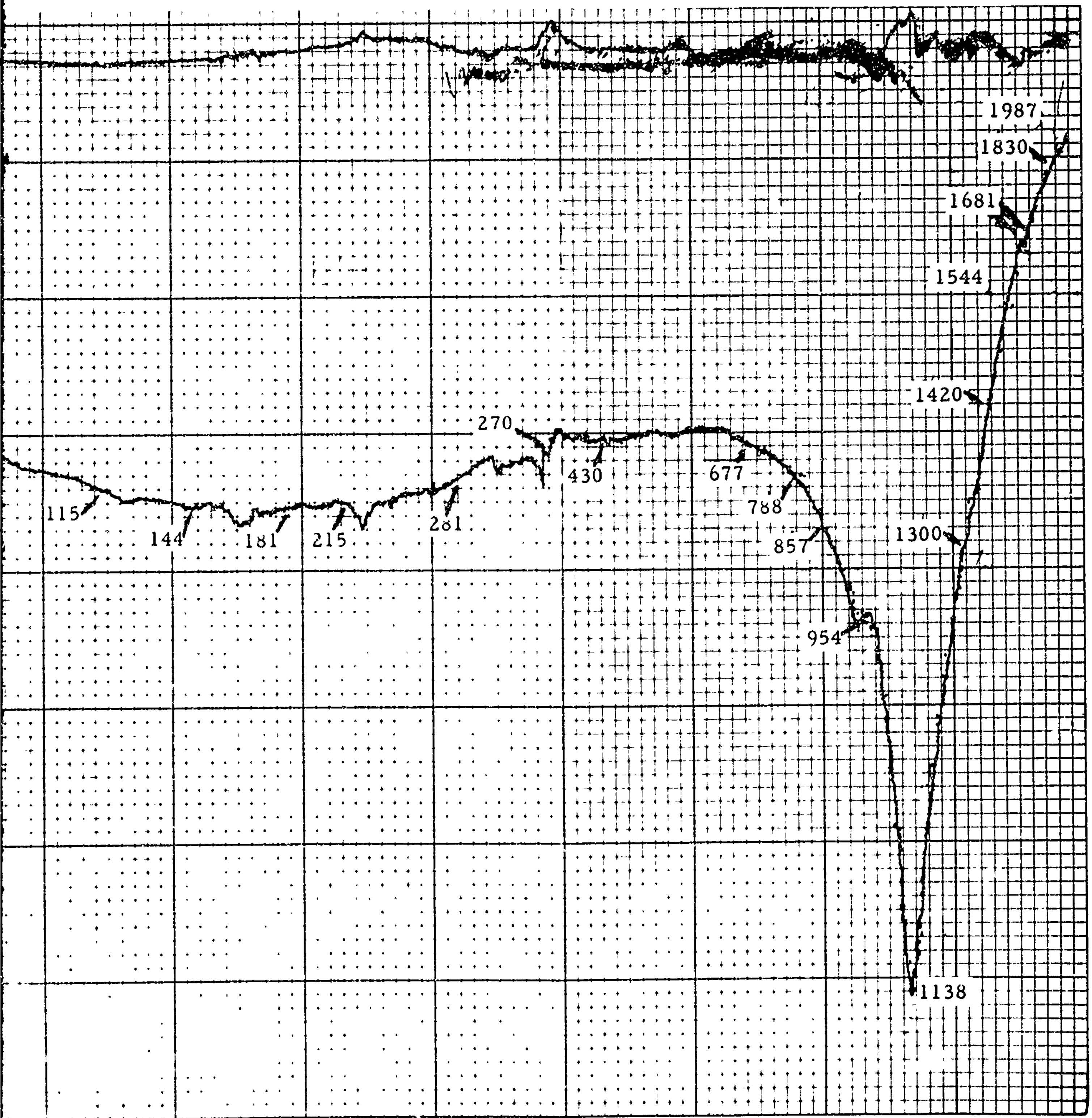
A



REQUENCY

Figure 3-95. Spacecraft No. 3 Resonant Responses, First Transverse Axis


AMPLITUDE OF INPUT POWER TO VIBRATION TABLE TO MAINTAIN CONSTANT G FORCE
INVERSE FUNCTION OF THE SPACECRAFT RESONANCES
115
144
FREQUENCY



QUENCY

Figure 3-96. Spacecraft No. 3 Resonant Responses, Second Transverse Axis

$$E_a \alpha_{REF} - \sigma \frac{T_{REF}^4 - T_w^4}{\frac{1}{\epsilon_{REF}} + \frac{A_{REF}}{A_w}\left(\frac{1}{\epsilon_w} - 1\right)} = 0, \qquad (3.19)$$

which includes the term $E_a \alpha_{REF}$ to account for radiation absorbed by the reference body from an external source. Terms not previously identified are:

A_w = area of wall surface of enclosure

A_{REF} = area of reference body

A_s = area of unknow body

E_a = radiation per unit area from external source

α_{REF} = absorptivity of reference body to external radiation

T_{REF} = temperature of surface of reference body

T_w = internal wall temperature of enclosure

ϵ_{REF} = emissivity of reference body in infrared region

ϵ_w = emissivity of internal wall surfaces of enclosure

3.13.4.3.3.3 For the body whose α/ϵ ratio is unknown, the same equation may be written, substituting α_s, T_s, A_s, and ϵ_s for the appropriate variables, thus:

$$E_a \alpha_s - \sigma \frac{T_s^4 - T_w^4}{\frac{1}{\epsilon_s} + \frac{A_s}{A_w}\left(\frac{1}{\epsilon_w} - 1\right)} = 0. \qquad (3.20)$$

Dividing (3.19) and (3.20) by α_{REF} and α_s, respectively, gives:

$$E_a = \sigma \frac{T_{REF}^4 - T_w^4}{\frac{\alpha_{REF}}{\epsilon_{REF}} + \alpha_{REF} \frac{A_{REF}}{A_w} \left(\frac{1}{\epsilon_w} - 1 \right)}, \text{ and} \tag{3.21}$$

$$E_a = \sigma \frac{T_s^4 - T_w^4}{\frac{\alpha_s}{\epsilon_s} + \alpha_s \frac{A_s}{A_w} \left(\frac{1}{\epsilon_w} - 1 \right)} \tag{3.22}$$

To eliminate E_a, divide (3.21) by (3.22):

$$1 = \frac{T_{REF}^4 - T_w^4}{\frac{\alpha_{REF}}{\epsilon_{REF}} + \alpha_{REF} \frac{A_{REF}}{A_w} \left(\frac{1}{\epsilon_w} - 1 \right)} \cdot \frac{\frac{\alpha_s}{\epsilon_s} + \alpha_s \frac{A_s}{A_w} \left(\frac{1}{\epsilon_w} - 1 \right)}{T_s^4 - T_w^4} \tag{3.23}$$

Multiplying by $\frac{\epsilon_s}{\alpha_s}$ and inverting:

$$\frac{\alpha_s}{\epsilon_s} = \frac{T_s^4 - T_w^4}{T_{REF}^4 - T_w^4} \cdot \frac{\alpha_{REF}}{\epsilon_{REF}} \cdot \frac{1 + \epsilon_{REF} \frac{A_{REF}}{A_w} \left(\frac{1}{\epsilon_w} - 1 \right)}{1 + \epsilon_s \frac{A_s}{A_w} \left(\frac{1}{\epsilon_w} - 1 \right)}$$

(3.24)

which is the desired relation.

3.13.4.3.3.4 To summarize, the tests were performed in three parts. First a black sphere ($\alpha/\epsilon = 1$) was placed in a vacuum chamber. (See figure 3-97.) The chamber was then evacuated to a pressure of 1×10^{-5} mm of Hg or better and the chamber walls cooled to -90°F. This provided a value for the average chamber wall temperature as seen by the spheres. Secondly, the black sphere was again rotated in the cooled chamber, but its surface was exposed to a high-intensity arc beam simulating the solar energy. (See figure 3-98.) The sphere's temperature was allowed to stabilize and then recorded to become the basis for comparison for all future spheres. Finally, the Geodetic Spacecraft was placed in the chamber (figure 3-99) and exposed to the arc while rotating. Once its temperature was stabilized, the value was recorded. Then, by solving the following equation, the desired α/ϵ ratio was determined.

$$\frac{\alpha}{\epsilon} = \frac{T_s^4 - T_w^4}{T_{REF}^4 - T_w^4} \quad \frac{\alpha_{REF}}{\epsilon_{REF}} \quad \frac{1 + \epsilon_{REF} \frac{A_{REF}}{A_w} \left(\frac{1}{\epsilon_w} - 1\right)}{1 + \epsilon_s \frac{A_s}{A_w} \left(\frac{1}{\epsilon_w} - 1\right)}$$

(3.25)

3.13.4.3.3.5 The α/ϵ was found to be 1.00 uncorrected and 0.97 with the correction factor included. The correction factor applied was that found previously for the chamber with a three foot sphere in it and adjusted for the smaller size of the Spacecraft. Tabel XX shows the stable temperatures for the black ball and satellite as well as the α/ϵ values.

3.13.4.3.4 Thermal-Vacuum Testing. The thermal-vacuum tests for Spacecraft No. 3 were conducted concurrently with the α/ϵ testing at the Applied Physics Laboratory during an 18-hour period on 18 August 1961. Light rings were installed in the vacuum chamber (figure 3-100) for temperature control and the Spacecraft was subjected to six temperature cycles using a thermocouple on the telemetering package as the controlling temperature. The specifications for this test called

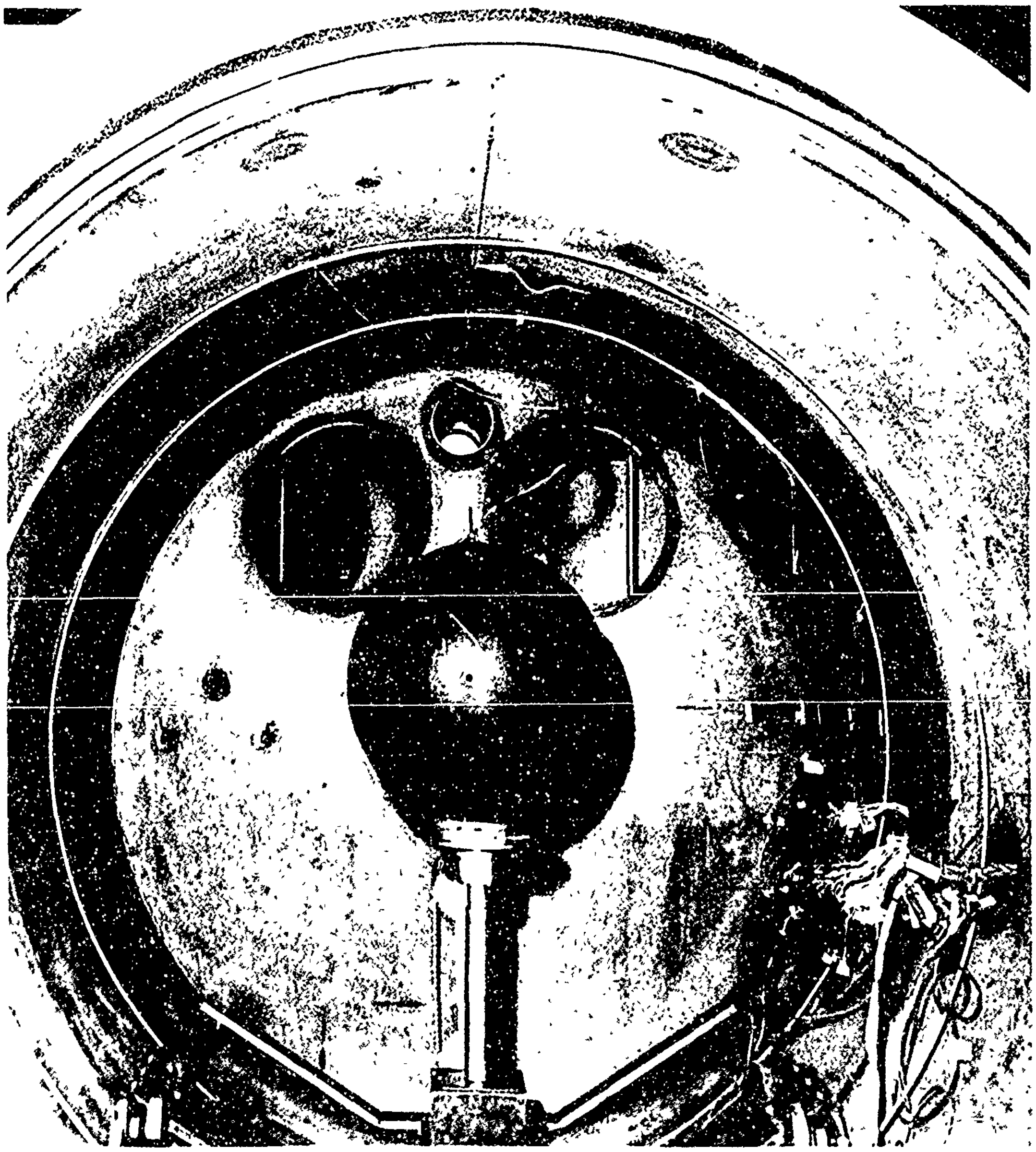


Figure 3-97. Black Sphere in Vacuum Chamber

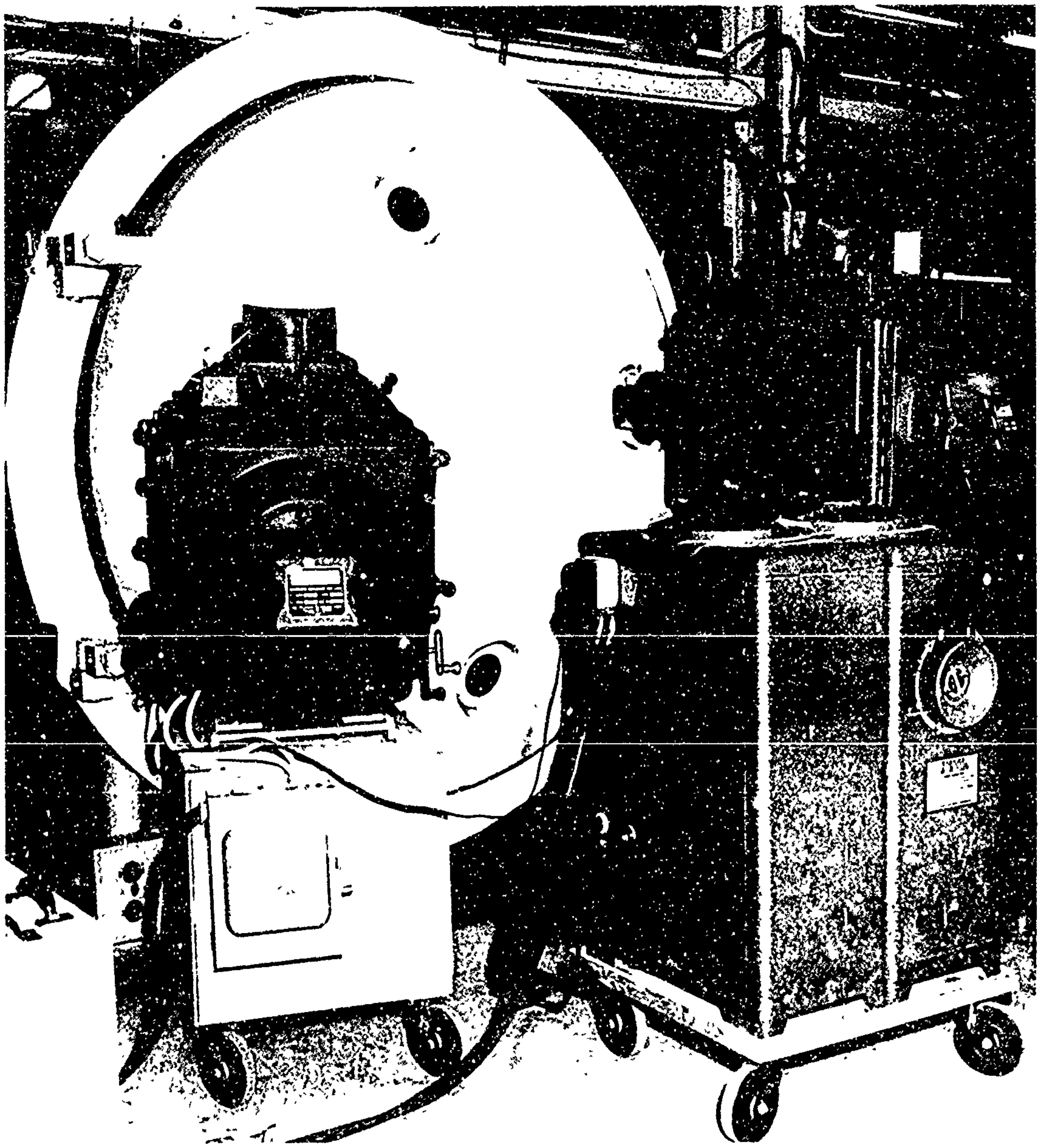


Figure 3-98. Arc Lights Used to Simulate the Sun

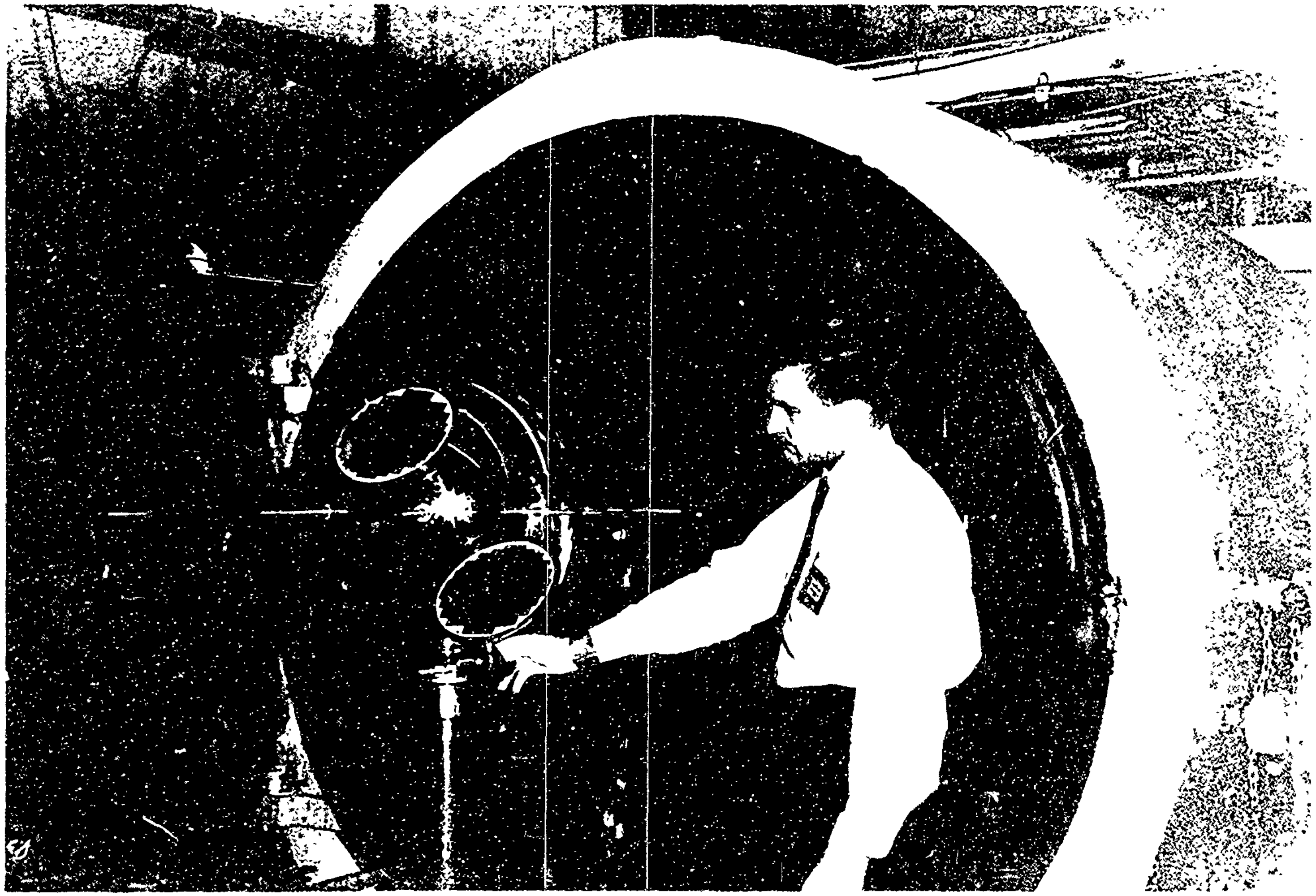


Figure 3-99. Geodetic Spacecraft Being Mounted in Vacuum Chamber

TABLE XX

STABLE TEMPERATURES AND A/ E VALUES

Thermocouple Location	Black Ball	SECOR	Uncorrected α/ϵ	Corrected α/ϵ
Top	$38^{o}F$	53.5	1.19	1.15
60^{o} up	50.5	54.5	1.04	1.04
30^{o} up	63	58.0	0.95	0.92
Equator	77	58.5	0.87	0.84
45^{o} down	68.5	58.5	0.94	0.91
Bottom	61	58.5	1.02	0.99
		Average	1.00	0.97
Chamber wall temperature = $-90^{o}F$				

for stabilizing the internal temperature of the satellite, (thermocouple on telemetering package), at $41^{o}F$ and then cycling the temperature about this value. Four 100-minute cycles were to be made about this temperature holding the telemetering package between $23^{o}F$ and $59^{o}F$. The temperature was then to be stabilized at $95^{o}F$ and four 100-minute cycles made holding the temperature between $77^{o}F$ and $113^{o}F$. Because of difficulty found in cooling the satellite, these specifications were adjusted by the project engineer in charge of the test and only six cycles were made. Three of these cycles were made about the lower temperature level and three about the upper temperature level, but the internal temperature was allowed to drift upward during the test so that the internal temperature was not held within the limits.

3.13.4.3.4.1 Table XXI is a list of the thermocouples. Figure 3-101 shows a plot of the

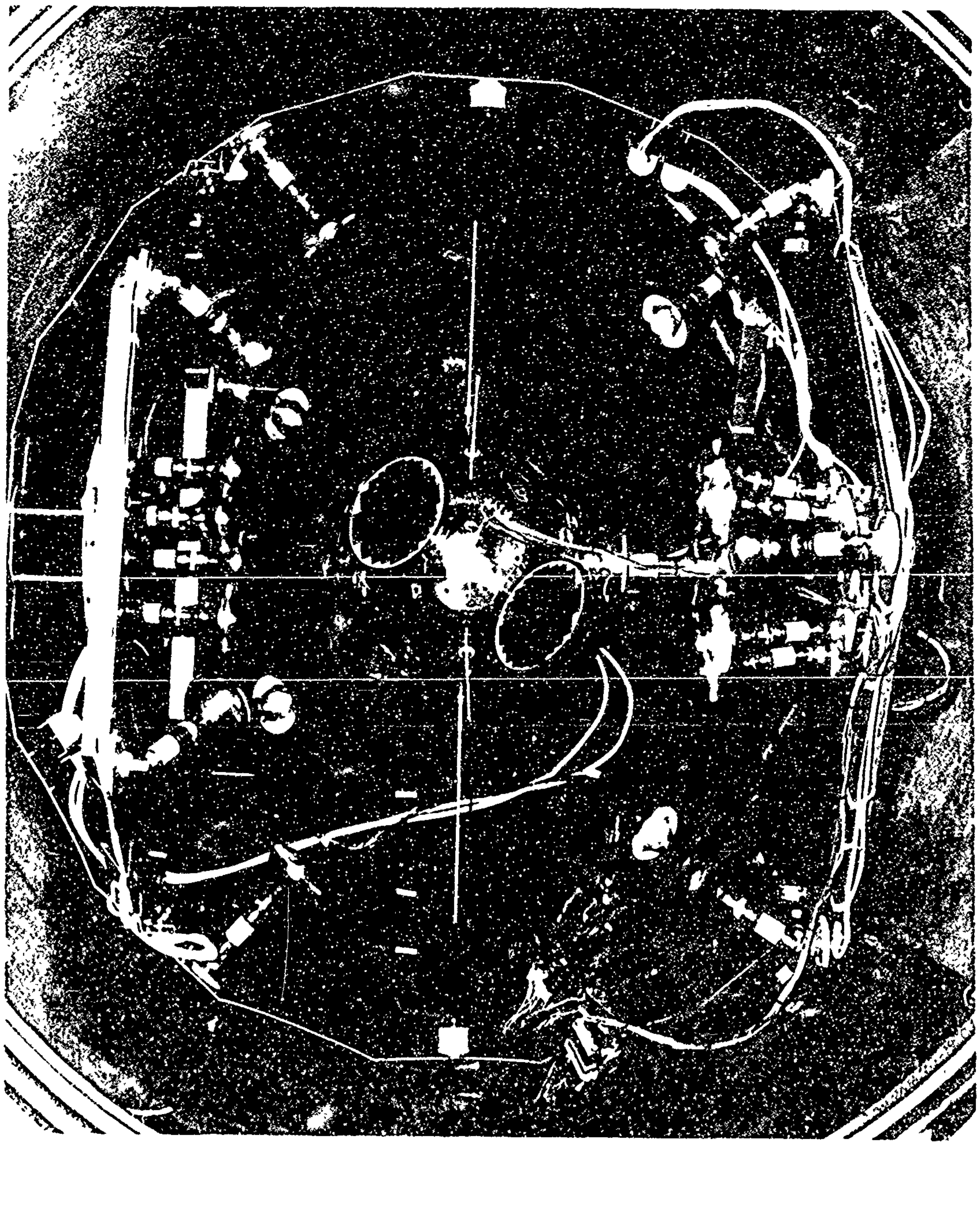


Figure 3-100. Geodetic Spacecraft in Chamber Showing Light Rings

TABLE XXI

THERMOCOUPLE IDENTIFICATION FOR THERMAL-VACUUM TEST
OF
GEODETIC SPACECRAFT NO. 3

Number	Location on the Specimen
1	Central tube top
2	Central tube at base of data amplifier
3	On telemetry board
4	Top inside skin
5	60° up inside skin
6	30° up inside skin
7	Chamber top
8	Solar cell middle
9	Chamber side wall
10	Chamber pressure
11	Central tube bottom
12	45° down inside skin
13	Equator inside skin
14	Equator outside skin
15	Bottom inside skin
16	Chamber bottom
17	Chamber back

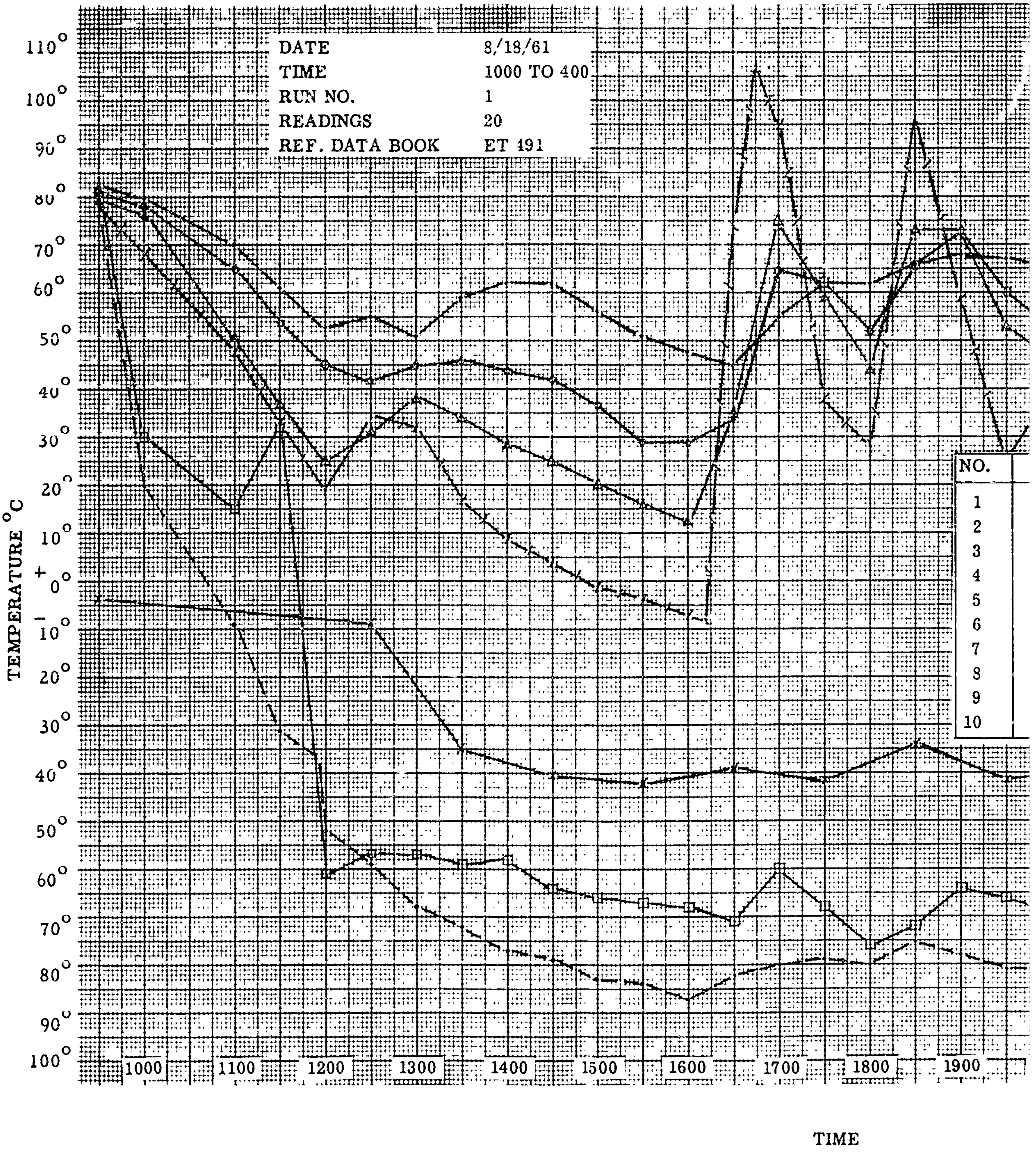
TABLE XXI (Cont)

Number	Location on the Specimen
18	Solar cell upper part
19	Flight plug outside skin
20	Chamber door

thermocouple responses during the thermal-vacuum test. The test data and original records for all these runs are on file in the APL Environmental Test Laboratory under ET 491. Telemetry data collected during the thermal-vacuum test of Spacecraft No. 3 is included in appendix C.

3.13.5 Conclusions. The only difficulty encountered during the test program was an intermittent stepping or channel-switching in the telemetry subsystem. This trouble was isolated to a printed circuit board and resulted in a redesign of the boards. Sporadic difficulty was encountered with the stepping of the redesigned system while conducting the second series of thermal-vacuum tests. As the system would later operate normally under the same conditions, it was believed that the difficulty was in the setup rather than in the Spacecraft. This conclusion was further supported by a thorough investigation of the Spacecraft upon completion of the test. The investigation showed that the Spacecraft was in perfect condition with no evidence of damage or equipment malfunction. However, as a precaution, some of the mechanical supports were insulated to prevent the 3V supply voltage from being shorted to ground.

3.13.5.1 The test setup for Spacecraft No. 2 was designed so that the difficulty could be isolated to either the Spacecraft or the setup. Difficulty with the stepping of the telemeter was encountered briefly on two occasions during the test. Each time the trouble was immediately traced to the setup. The difficulty proved to be a result of stray pulses being picked up by the long interconnecting cables between the vacuum chamber and the test bench. The pulses were caused by electro-mechanical devices such as motors, or shakers, operating in the test area. The stray pulses were coupled into


DATE 8/18/61
TIME 1000 TO 400
RUN NO. 1
READINGS 20
REF. DATA BOOK ET 491
TEMPERATURE °C
110° 100° 90° 80° 70° 60° 50° 40° 30° 20° 10° + 0° - 10° 20° 30° 40° 50° 60° 70° 80° 90° 100°
1000 1100 1200 1300 1400 1500 1600 1700 1800 1900
NO.
1 2 3 4 5 6 7 8 9 10
TIME

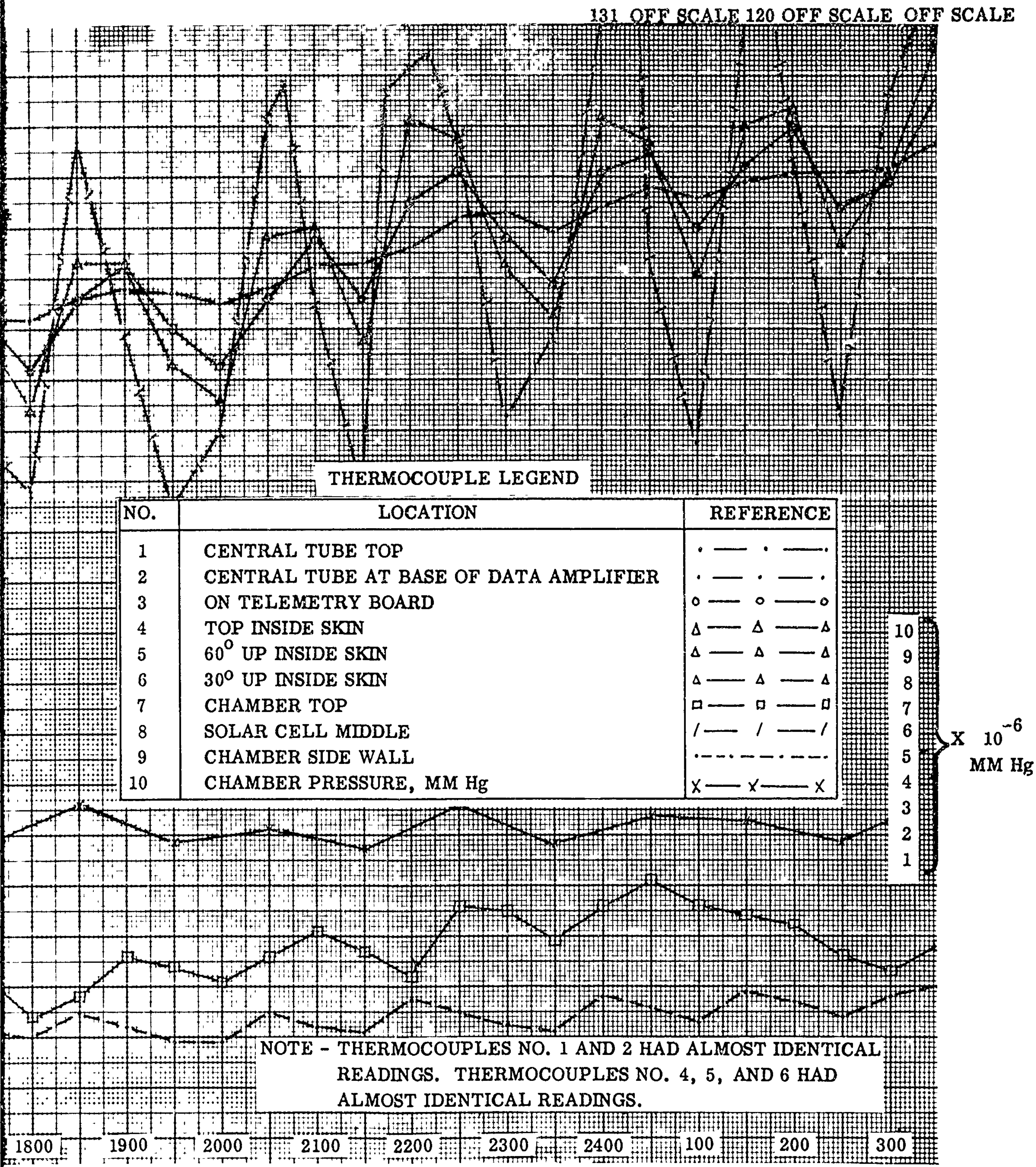
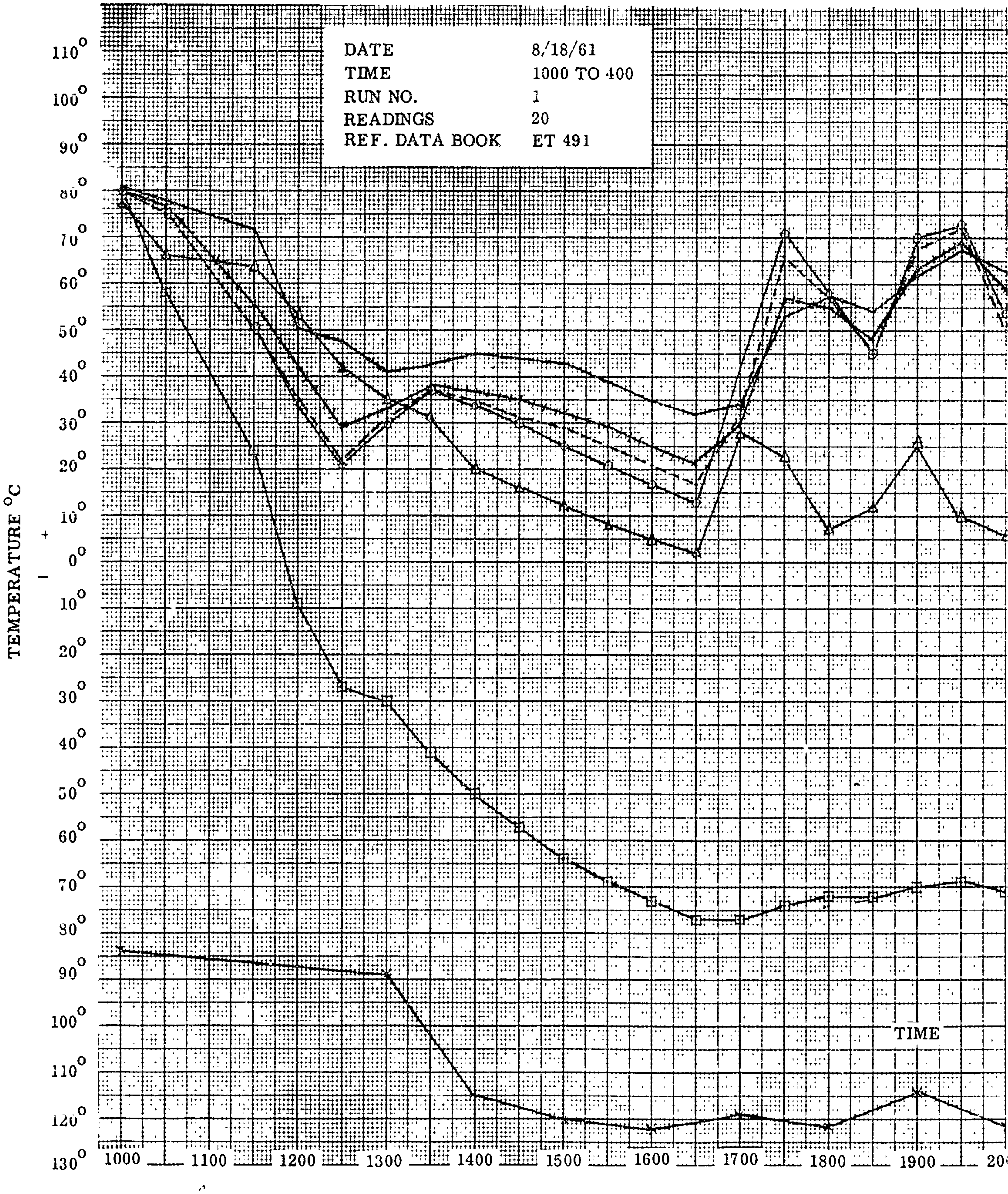


Figure 3-101. Thermocouple Responses for Geodetic Spacecraft No. 3 Thermal-Vacuum Tests (Sheet 1 of 3)


DATE 8/18/61
TIME 1000 TO 400
RUN NO. 1
READINGS 20
REF. DATA BOOK ET 491
TEMPERATURE °C
+
−
110°
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
100°
110°
120°
130°
1000
1100
1200
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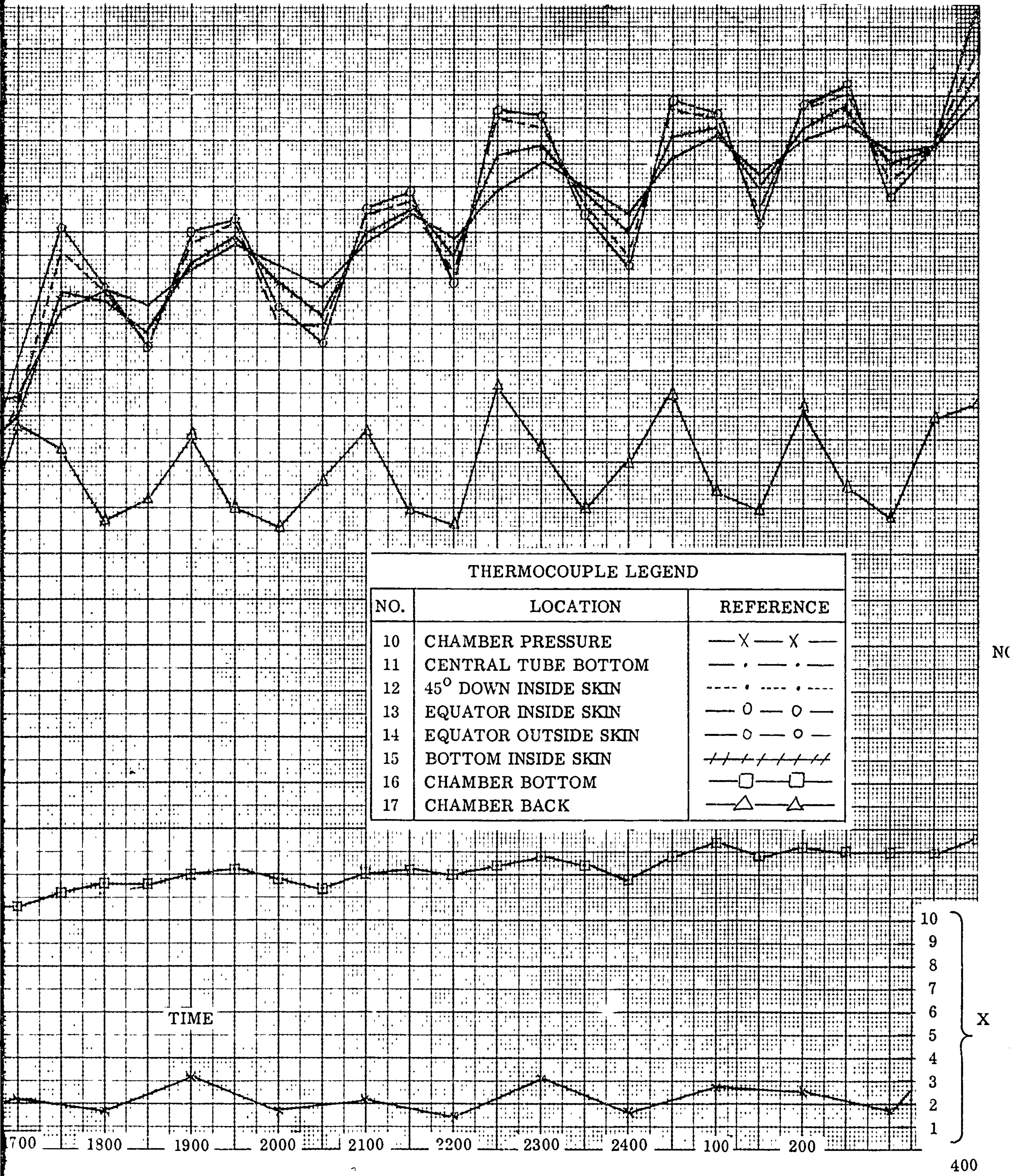
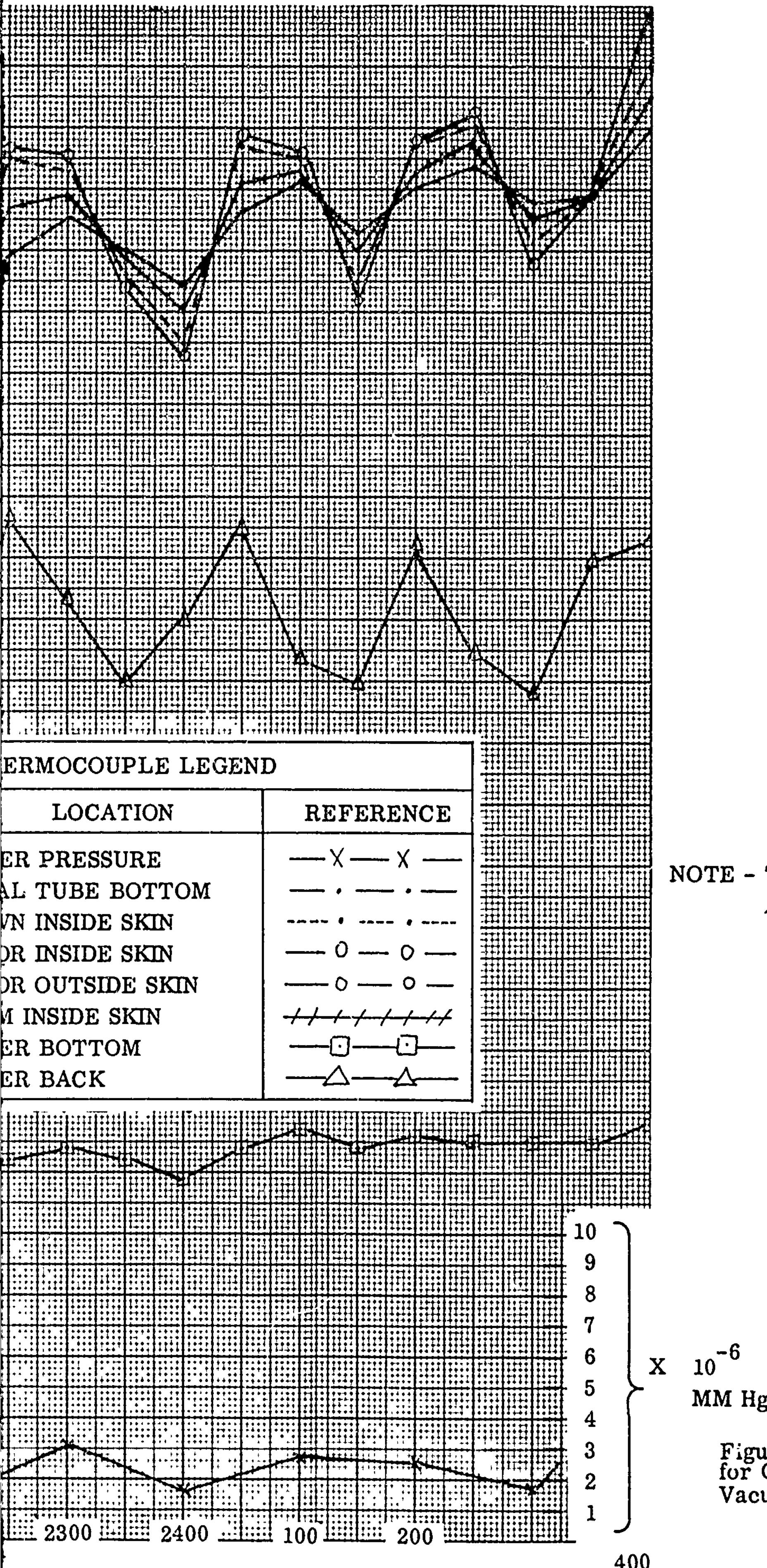
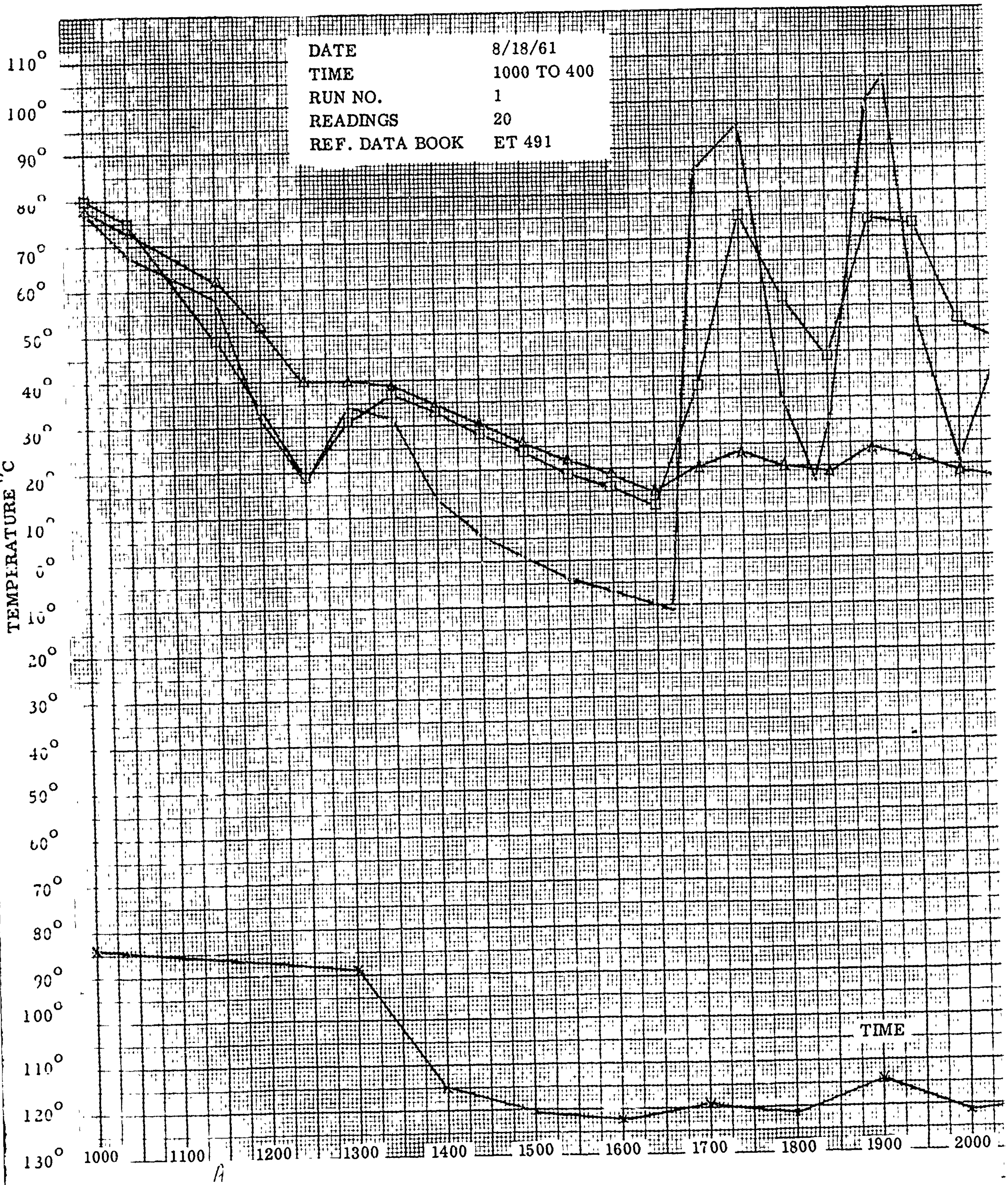
1400
1500
1600
1700
1800
1900
TIME


THERMOCOUPLE LEGEND
NO. LOCATION REFERENCE
10 CHAMBER PRESSURE
11 CENTRAL TUBE BOTTOM
12 45° DOWN INSIDE SKIN
13 EQUATOR INSIDE SKIN
14 EQUATOR OUTSIDE SKIN
15 BOTTOM INSIDE SKIN
16 CHAMBER BOTTOM
17 CHAMBER BACK
TIME
1700 1800 1900 2000 2100 2200 2300 2400 100 200
400
10 9 8 7 6 5 4 3 2 1
X
N



NOTE - THERMOCOUPLES NO. 13 AND 14 HAD ALMOST IDENTICAL READINGS.

Figure 3-101. Thermocouple Responses for Geodetic Spacecraft No. 3 Thermal-Vacuum Tests (Sheet 2 of 3)


DATE 8/18/61
TIME 1000 TO 400
RUN NO. 1
READINGS 20
REF. DATA BOOK ET 491
TEMPERATURE °C
110°
100°
90°
80°
70°
60°
50°
40°
30°
20°
10°
0°
10°
20°
30°
40°
50°
60°
70°
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TIME

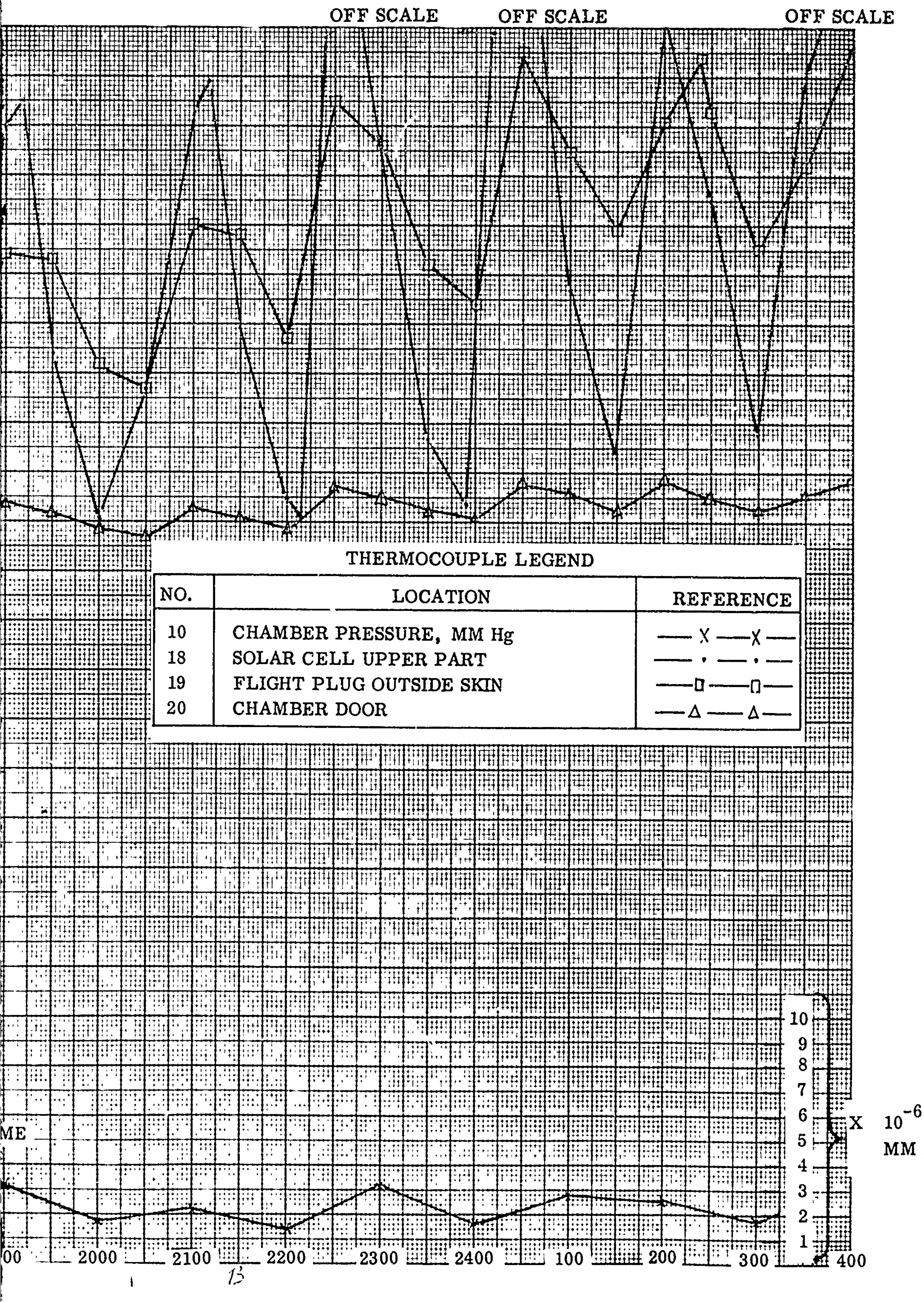


Figure 3-101.
for Geodetic S
Vacuum Tests

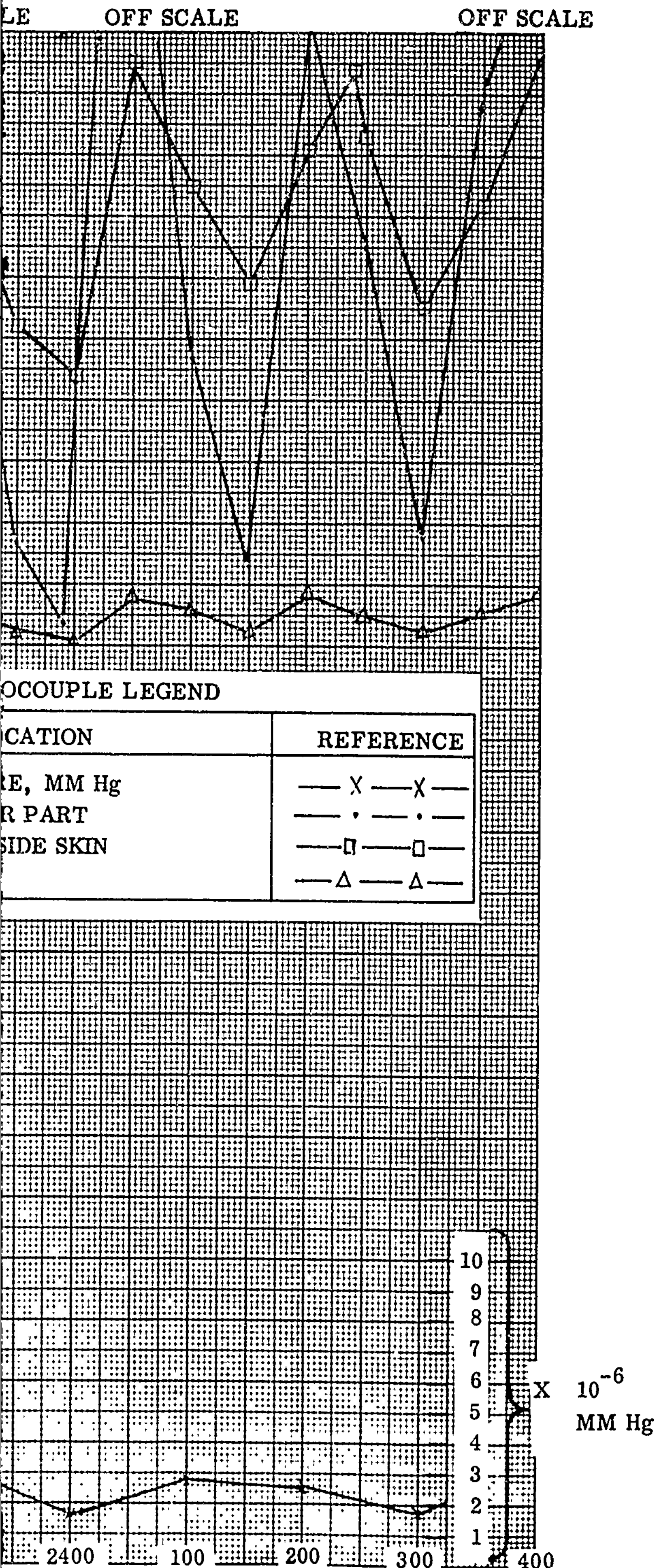


Figure 3-101. Thermocouple Responses for Geodetic Spacecraft No. 3 Thermal-Vacuum Tests (Sheet 3 of 3)

the circuit when the test panel stepping switch was closed, effectively preventing stepping control. Thus the commutator could not be made to stop on the desired channel. The pulses are easily visible on an oscilloscope. The system functioned normally as soon as the source of the interference was removed. No difficulty was encountered with the Spacecraft itself during the entire environmental test program.

3.14 Calculation of Spacecraft Mass Constants. Integration of the Spacecraft to its launch vehicle required the calculation of its mass distribution, center of gravity, moment of inertia, shear and flexural rigidity. The following data pertains only to Spacecraft No. 1, the prototype model. The measurements were not made on the two flight Spacecraft, as the expected deviations from the prototype data were not considered sufficient to warrant their calculation.

3.14.1 Mass Distribution. The mass distribution diagram (figure 3-102) was calculated from the weights of the various spacecraft components and their relative position within the shell.

3.14.2 Center of Gravity. Data for the center of gravity was obtained by suspending the Spacecraft from an antenna mount in X and Y axis and from the separation fixture in the Z axis. A fixture with spirit levels was placed over the support wire and moved about its axis until it was level in all directions. Center of gravity lines were determined by measuring the movement of the fixture from its initial axis. The center of gravity proved to be on the Z axis 3/4-inch below the equator of the Spacecraft.

3.14.3 Moment of Inertia. For the moment of inertia calculations the Spacecraft was suspended from a precalibrated wire in each of its three axes in succession. During each suspension, the Spacecraft was given six trials to determine the period of revolution. In each trial the Spacecraft was given one revolution and the time for five periods was noted. The individual readings were averaged and divided by 5 to give the period for each axis. Calculations were as follows:

$$I = 0.1276T^2 - 0.2812 - IF \qquad (3.26)$$

where: 0.1276, 0.2812 are constants of the test setup, and IF = moment of inertia for fixture.

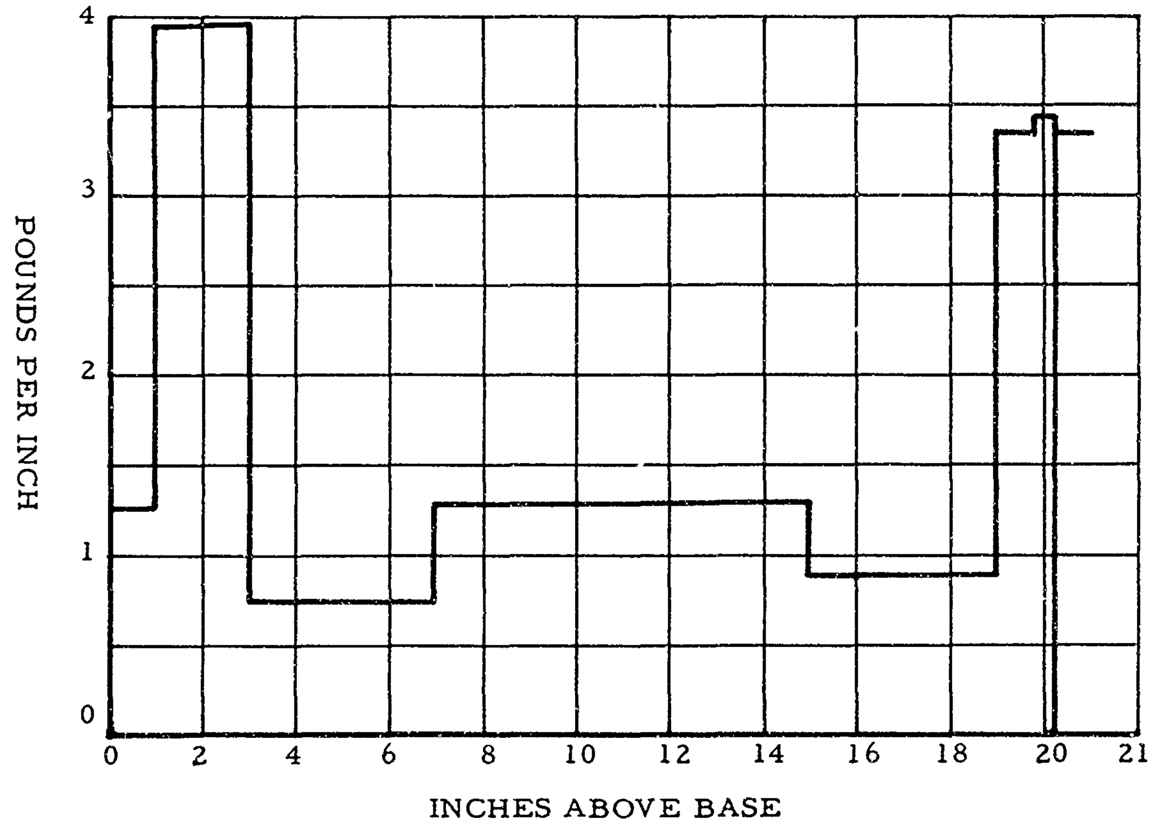


Figure 3-102. Mass Distribution Diagram for Prototype Spacecraft

$IF = 0.197$ for the X axis

$= 0.197$ for the Y axis

$= 2.047$ for the Z axis

$T = 21.868$ seconds for the X axis

$= 21.756$ seconds for the Y axis

$= 18.56$ seconds for the Z axis

Then: $I_x = 0.1267\,(21.868)^2 - 0.2812 - 0.197$

$= 0.418$ slug ft^2

$I_y = 0.1267\,(21.756)^2 - 0.2812 - 0.197$

$= 0.412$ slug ft^2

$I_z = 0.1267\,(18.56)^2 - 0.2812 - 2.047$

$= 0.287$ slug ft^2

3.14.4 Shear and Flexural Rigidity. Shear and flexural rigidity diagrams are shown in figures 3-103 and 3-104, respectively.

3.15 Spacecraft Mockup. Cubic fabricated a mockup of the Geodetic Spacecraft which was used in separation tests during development of the Composite I Multiple Satellite structure. (See figure 3-105.)

3.16 Hoisting Sling. Cubic has designed a unique hoisting sling for handling the Spacecraft during the prelaunch phase of the program. The sling consists of an inner protective cover and an outer nylon net. The inner cover protects the spacecraft's silicon monoxide thermal coating and the net serves to support the Spacecraft when it is lifted into position on the launch vehicle. A hole in the cover and the net permits access to the separation fitting. Thus the Spacecraft

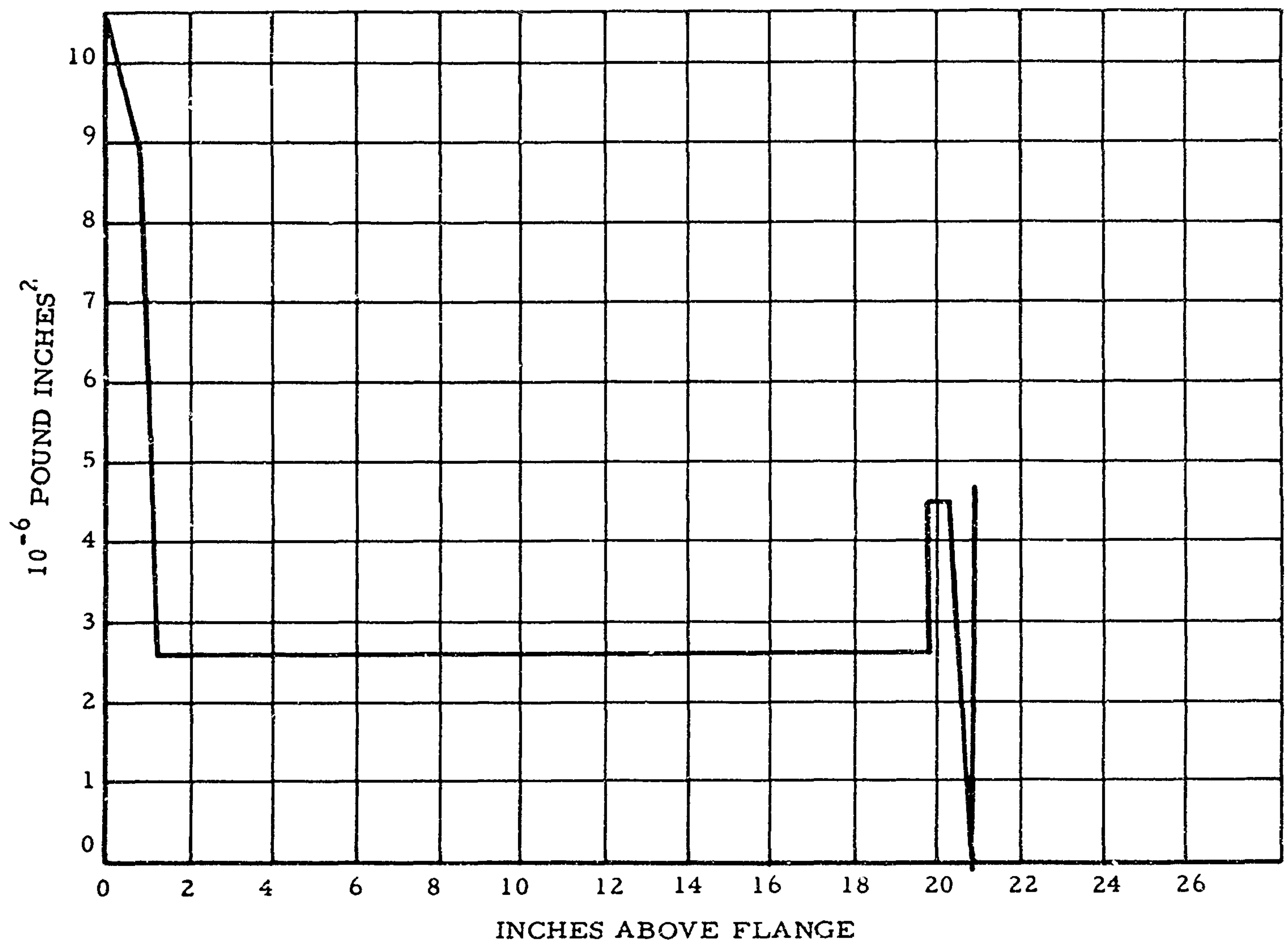


Figure 3-103. Shear Rigidity Diagram for Prototype Spacecraft

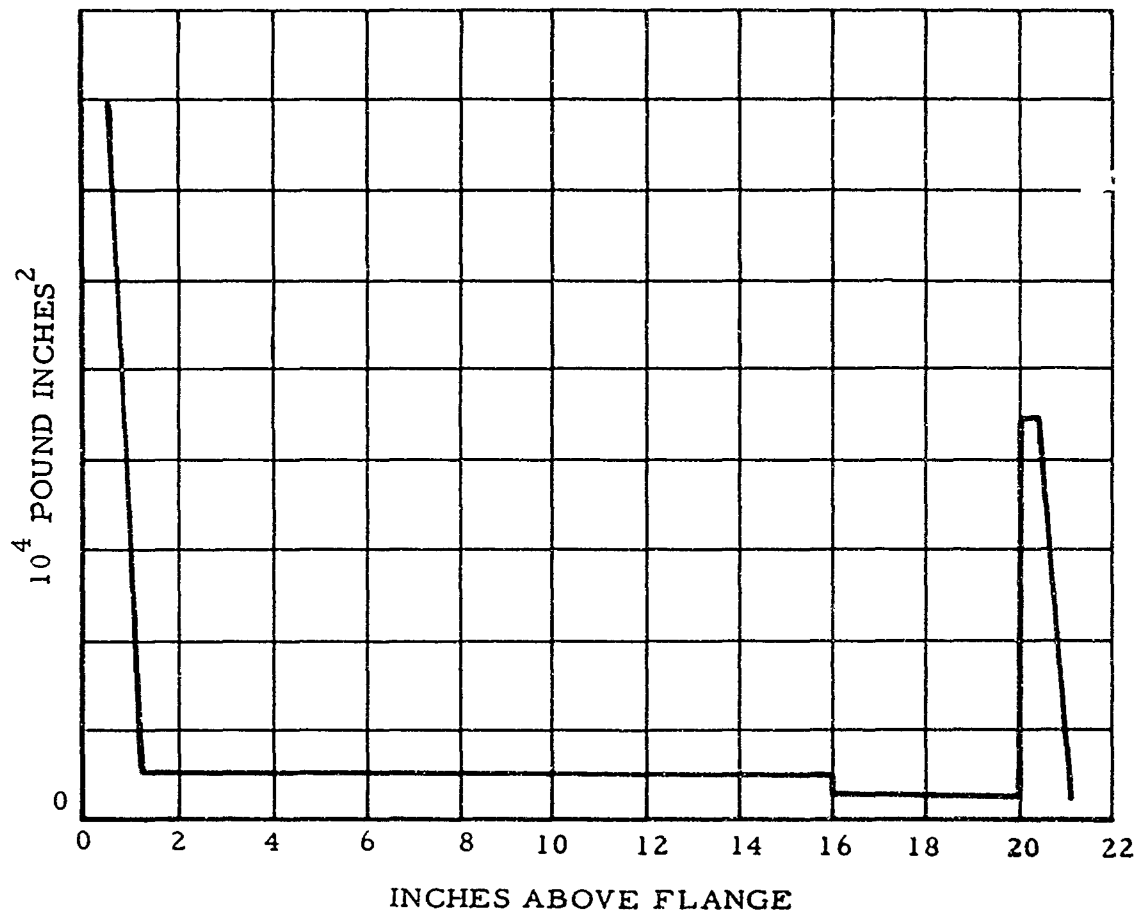


Figure 3-104. Flexural Rigidity Diagram for Prototype Spacecraft

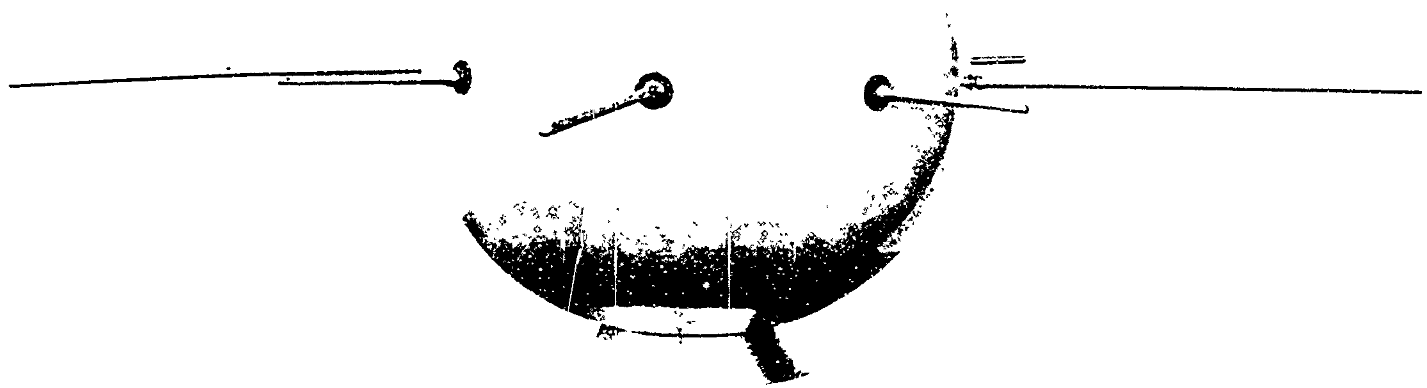


Figure 3-105. Geodetic Spacecraft Mockup

may be supported by the sling while it is clamped to the launch vehicle. The use of the hoisting sling provides an efficient and inexpensive means of handling the Spacecraft precluding the need for external hoisting fixtures.

3.17. Special Spacecraft Test Equipment. Cubic designed and constructed three special test devices to be utilized for checking out the Spacecraft during prelaunch operations. (See figure 3-106.) This equipment consists of a test transmitter, telemetry test receiver and a test panel. It is employed in conjunction with standard test devices to check operation of the transponder and to monitor the telemetry output. The following paragraphs describe each unit and provide a brief disscussion of operation.

3.17.1 Test Panel. The test panel provides the external switches and interconnections required to monitor the telemetry channels through the spacecraft flight plug. External power, furnished to the Spacecraft through the test panel, may be utilized to operate the electronic subsystems and charge the batteries. Two Simpson 260 voltmeters (interconnected with the test panel) provide visual voltage and current measurements of the telemetry commutator channels. Table XXII lists the function of the test panel controls and figure 3-107 shows the front of the panel and the location of the controls. Figure 3-108 is a wiring diagram of the test panel.

3.17.2 Test Transmitter. The test transmitter simulates a SECOR ground station and is utilized to checkout the Geodetic Spacecraft transponder during prelaunch operations. The transmitter consists of a crystal oscillator rf generator, phase modulator, reactive X8 multiplier, buffer amplifier, band-pass filter, and four crystal oscillator modulation circuits. The rf generator output (52.617 mc) is multiplied eight times to provide the 420.9375 carrier. The four crystal oscillator modulation circuits provide the carrier modulation required to checkout the transponder select call, telemetry on/off and data loop functions. External switches on the test transmitter connect the individual modulation circuits to the phase modulator. External connectors permit separate monitoring of the carrier and the modulation (or injection of an externally generated modulation signal). In normal operation, a $\lambda/2$ dipole is connected to the carrier jack. The transmitter operates from a 12V external power source and draws 54 ma of current (maximum). Table XXIII lists the function of the test transmitter controls and connectors, and figure 3-109 shows their location.

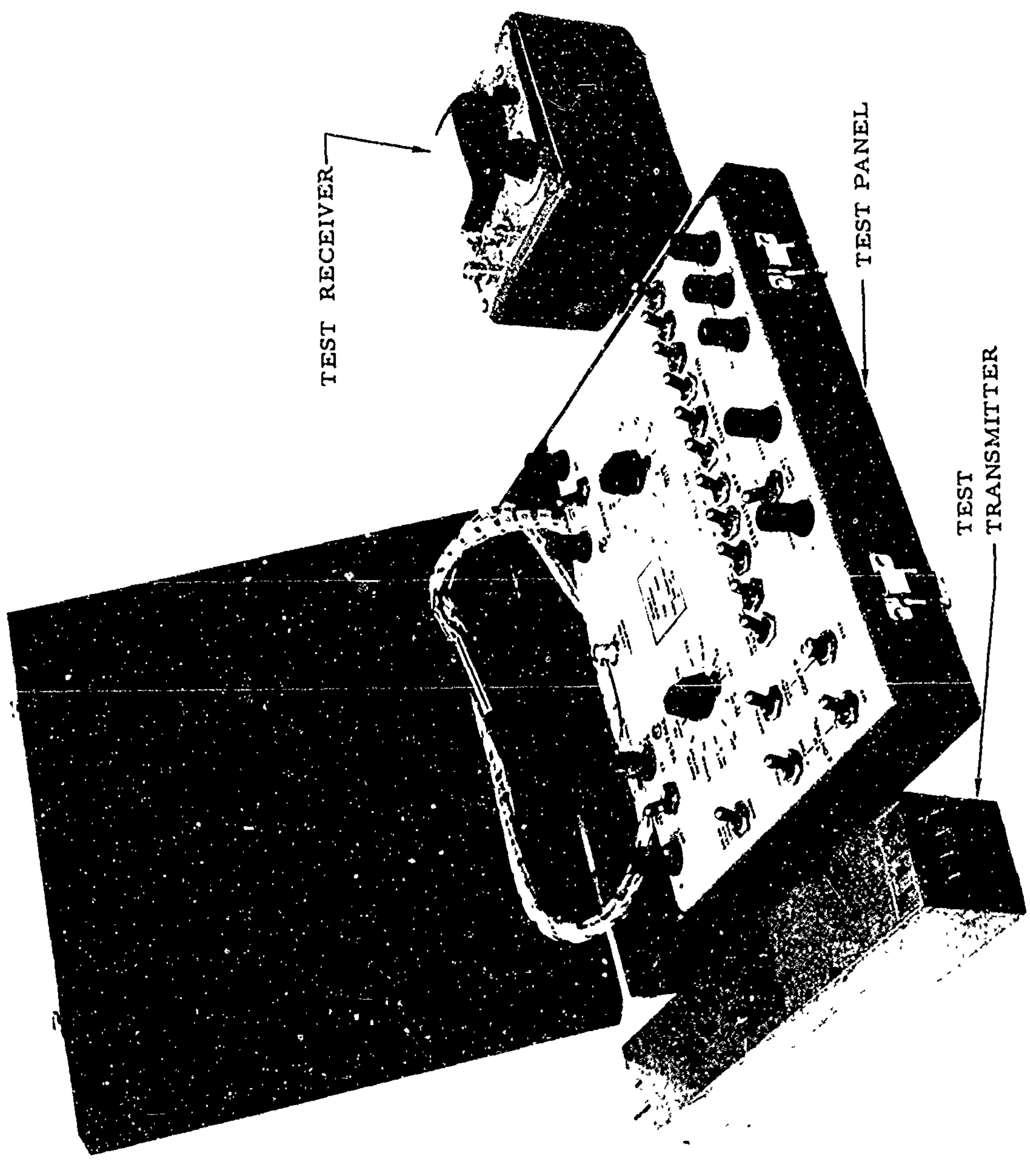


Figure 3-106. Special Test Equipment for Geodetic Spacecraft

TABLE XXII

TEST PANEL SWITCH FUNCTIONS

Switch	Function
VOLTMETER	Provides (+) and (-) terminals for connecting a Simpson 260 voltmeter to the panel. ON/OFF switch connects voltmeter into circuit indicated by VOLTAGE selector switch.
TELEMETRY MODULATION	Provides a coaxial connector for sampling telemetry modulation.
CURRENT METER	Provides (+) and (-) terminal for connecting a Simpson 260 voltmeter to the panel. ON/OFF switch connects voltmeter into circuit indicated by CURRENT selector switch.
SOLAR CELLS	Provides switch which connects solar cell output to the VOLTAGE and CURRENT selector switches.
SHORT	When the switch is in SHORT position ~~normal~~ solar cells are shorted to ground in order to measure ~~spacecraft load is removed from solar cell output.~~ short circuit current.
~~NORMAL~~ LOAD	When switch is in ~~NORMAL~~ LOAD position ~~normal~~ solar cells open-circuit voltage or charging current ~~spacecraft load is in solar cell output.~~ may be measured
VOLTAGE	The selector switch connects the various indicated functions to the voltmeter for monitoring.
CURRENT	The selector switch connects the various indicated functions to the current meter for monitoring.
TM CONTROL	Provides switches for controlling the telemeter externally.

TABLE XXII (Cont)

Switch	Function
NORMAL	This position permits the transponder to energize the telemetry subsystem via the telemetry call channel.
BYPASS	This position removes the telemetry call channel control, and permits the test panel operator to energize the telemetry subsystem manually.
ON/OFF	When above switch is in BYPASS, ON/OFF switch enables operator to control power to telemetry subsystem.
COM'T'R	Enables test panel operator to permit commutator to run normally, or to step to individual channels manually.
NORMAL	When switch is in this position, commutator steps automatically.
MANUAL STEP	When switch is in this position, operator may step commutator manually by pressing STEP button.
CIRCUIT OFF or CURRENT MONITOR	When any of the series of switches located above this designation are in the CIRCUIT OFF position, any power connected to the indicated circuit will pass through the CURRENT METER.
CIRCUIT ON	When any of the switches located below this designator are in the CIRCUIT ON position, power is connected directly to the indicated circuit, by-passing the CURRENT METER.

TABLE XXII (Cont)

Switch	Function
EXT. 12V SUPPLY	Provides (+) and (-) terminals for connecting external power to the spacecraft electronic subsystems.
12V SUPPLY	Provides a switch for selecting either INTERNAL or EXTERNAL power.
BATT. NO. 1 CHARGE, GROUND BATT. NO. 2 CHARGE	Provides the terminals for connecting external power to the battery charge circuits.

3.17.3 Telemetry Test Receiver. The test receiver is a self-powered superheterodyne device that is utilized to verify the presence of the telemetry signal during checkout. Sensitivity is approximately -50 to -60 dbm. Operating power (13.5V at 33 ma) is furnished by two dry cell batteries connected in series. Table XXIV lists the function of the controls and figure 3-110 shows location of the controls on the front panel.

3.18 Spacecraft Shipping Containers. Figure 3-111 and 3-112 show the details of the shipping containers which Cubic designed and built for the Geodetic Spacecraft.

3.19 Final Calibration of Telemetry Channels. Prior to delivery of the spacecraft to the launch site, Cubic performed a final calibration of the telemetry channels. A typical strip-chart recorded during the tests conducted on 21 November is reproduced in figure 3-113.

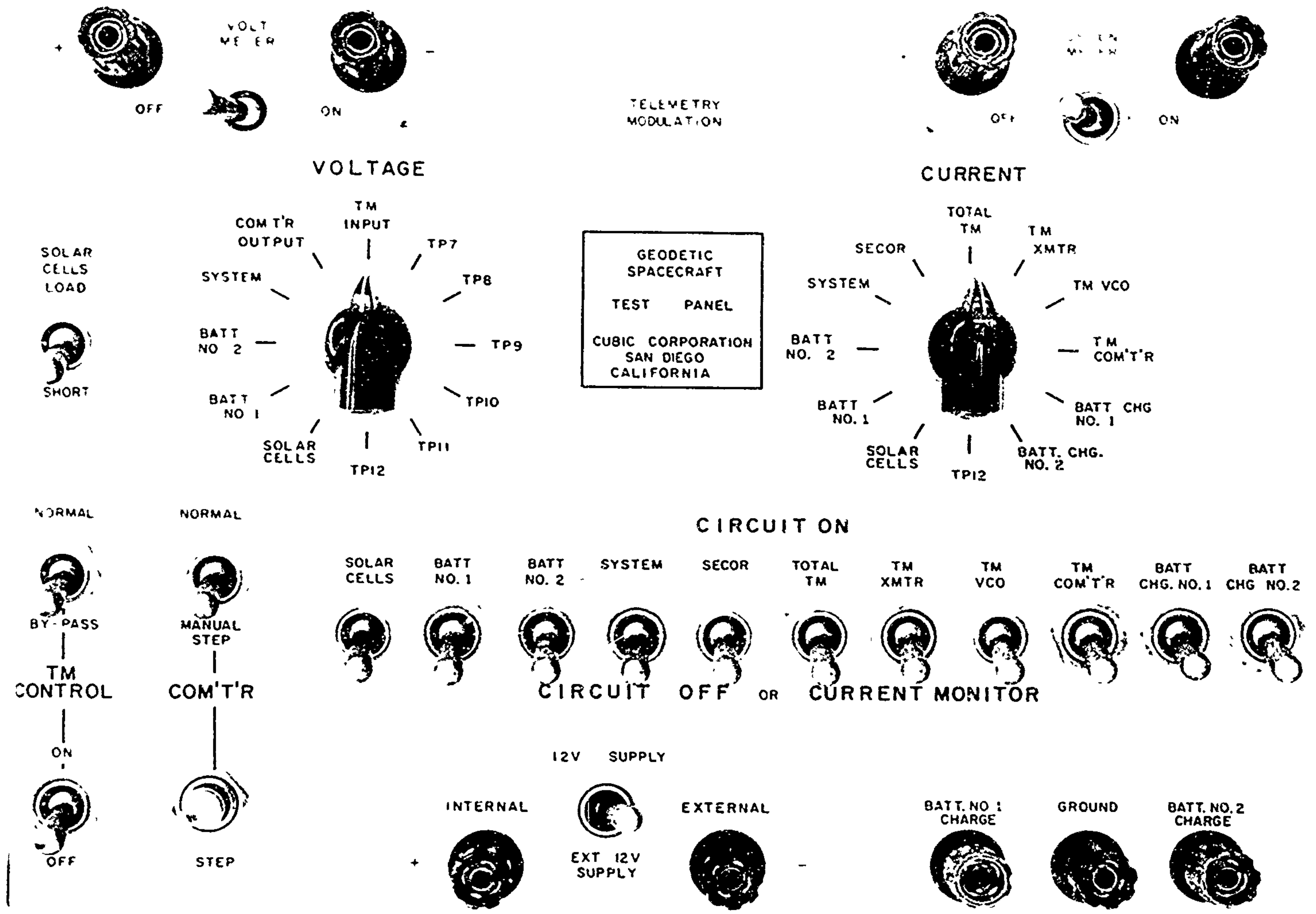
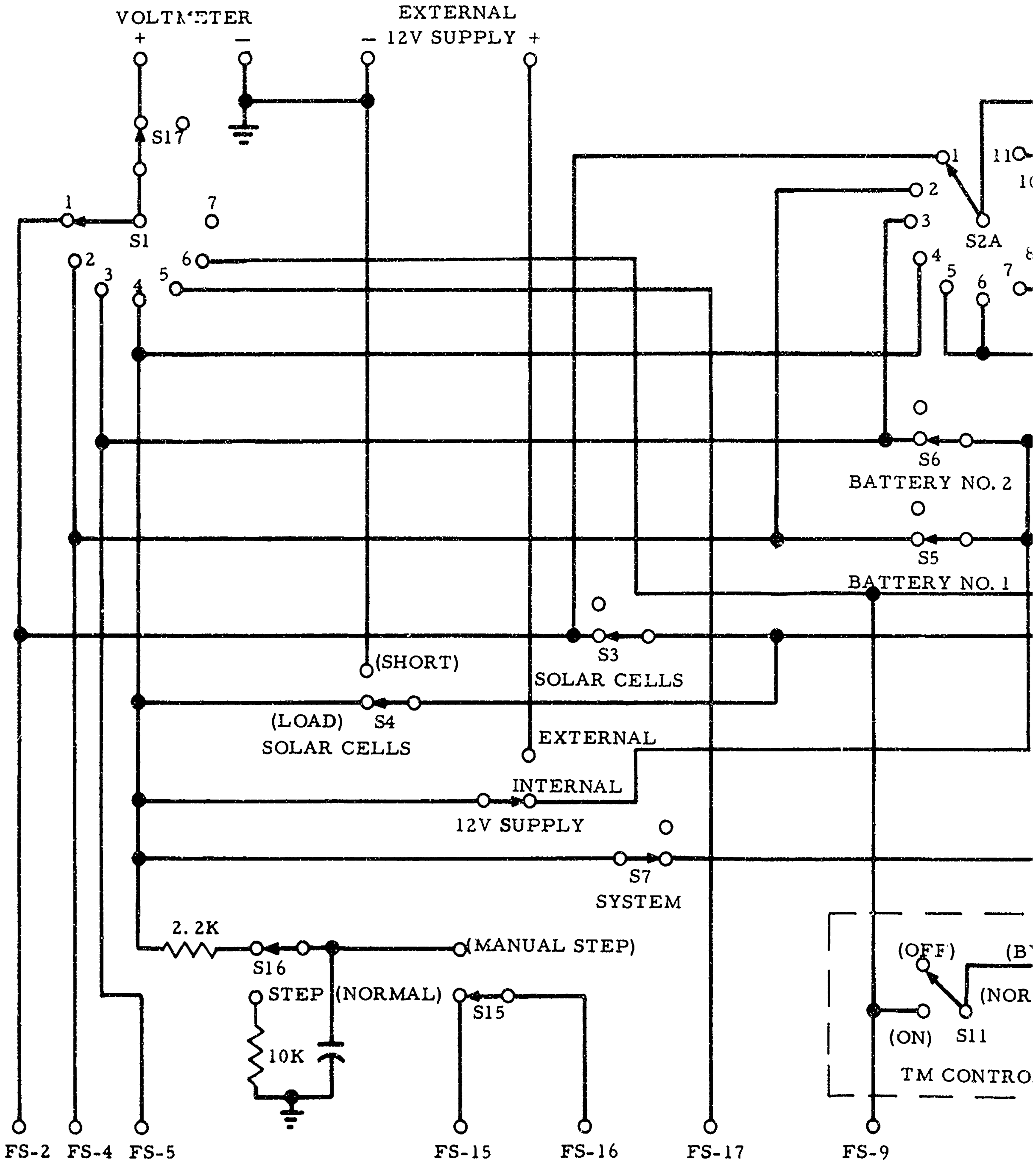
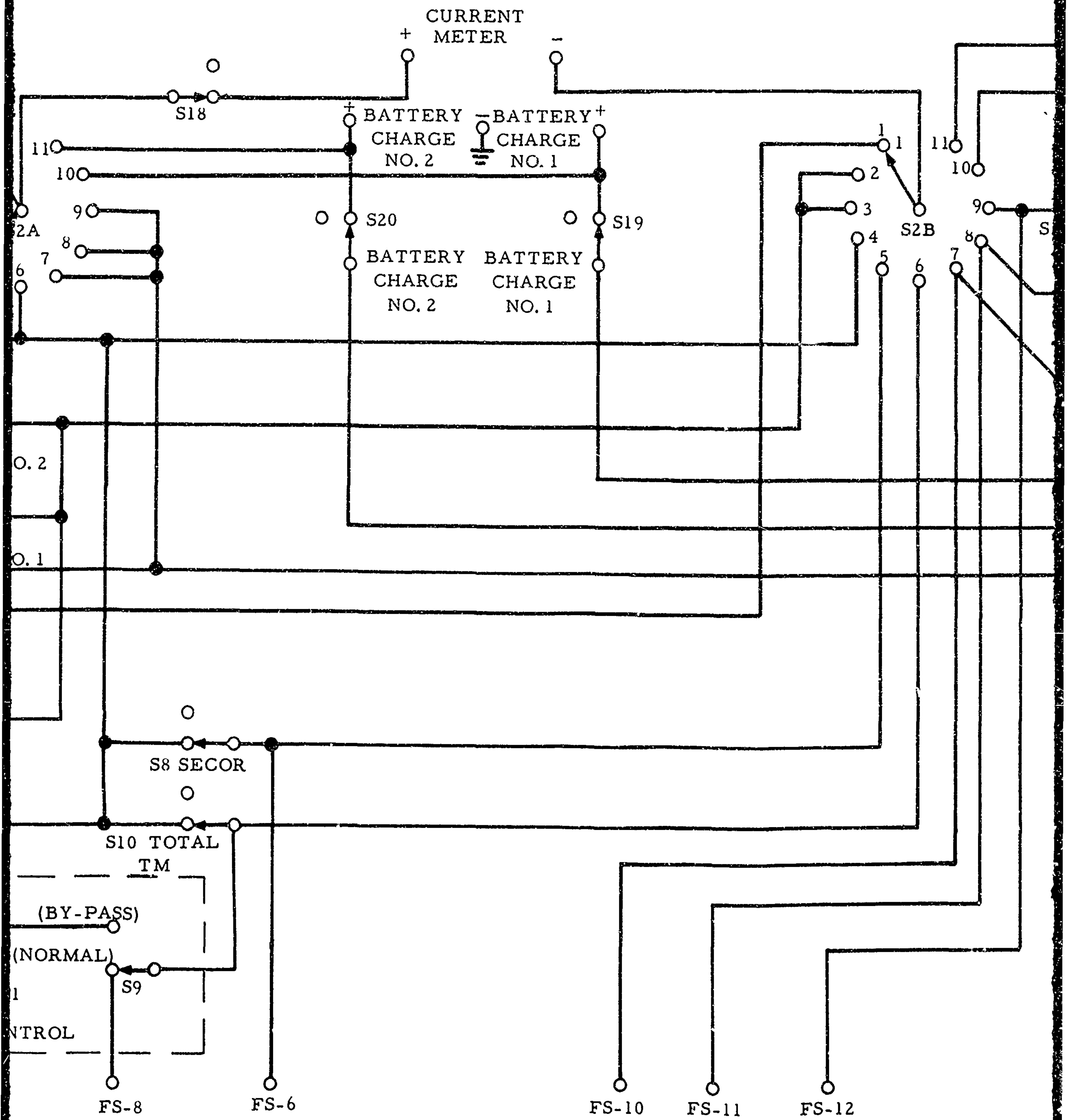


Figure 3-107. Front Panel of Test Panel


VOLTMETER
+
−
EXTERNAL
− 12V SUPPLY +
S17
S1
S2A
S6
BATTERY NO. 2
S5
BATTERY NO. 1
S3
SOLAR CELLS
(SHORT)
(LOAD)
S4
SOLAR CELLS
EXTERNAL
INTERNAL
12V SUPPLY
S7
SYSTEM
2.2K
S16
(MANUAL STEP)
STEP
(NORMAL)
S15
10K
(OFF)
(ON)
S11
TM CONTRO
FS-2
FS-4
FS-5
FS-15
FS-16
FS-17
FS-9



CURRENT
METER
S18
BATTERY
CHARGE
NO. 2
BATTERY
CHARGE
NO. 1
S20
S19
BATTERY
CHARGE
NO. 2
BATTERY
CHARGE
NO. 1
2A
S2B
O. 2
O. 1
S8 SECOR
S10 TOTAL
TM
(BY-PASS)
(NORMAL)
S9
NTROL
FS-8
FS-6
FS-10
FS-11
FS-12

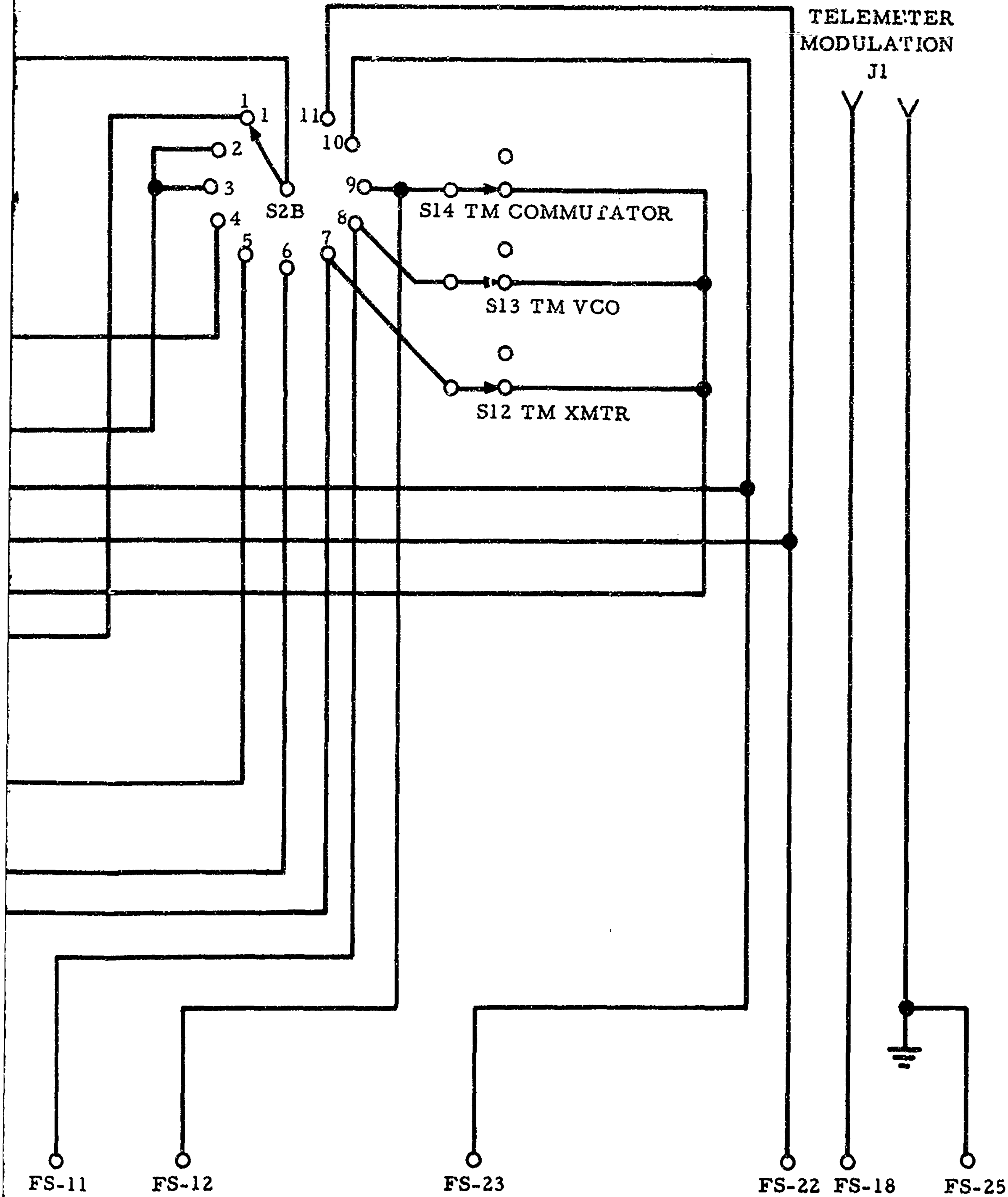


Figure 3-108. Wiring Diagram of Test Panel.

TABLE XXIII

TEST TRANSMITTER CONTROL AND CONNECTOR FUNCTION

Control or Connector	Function
CARRIER	Energizes the rf oscillator providing 420.9375 mc at CARRIER output connector
SELECT CALL	Energizes the select call oscillator providing 561.000 kc phase modulation of the carrier
TELEMETRY ON	Energizes the telemetry on oscillator providing 575.000 kc phase modulation of the carrier
TELEMETRY OFF	Energizes the telemetry off oscillator providing 570.000 kc phase modulation of the carrier
MODULATION	Permits monitoring of any energized modulation oscillator
DATA (located beneath the power input connector)	Energizes the very fine data oscillator providing 585.533 kc phase modulation of the carrier
POWER INPUT	Provides terminals for connecting external 12V power and ground
CARRIER	Permits monitoring of the carrier and provides antenna input

Figure 3-109. Front Panel of Test Transmitter

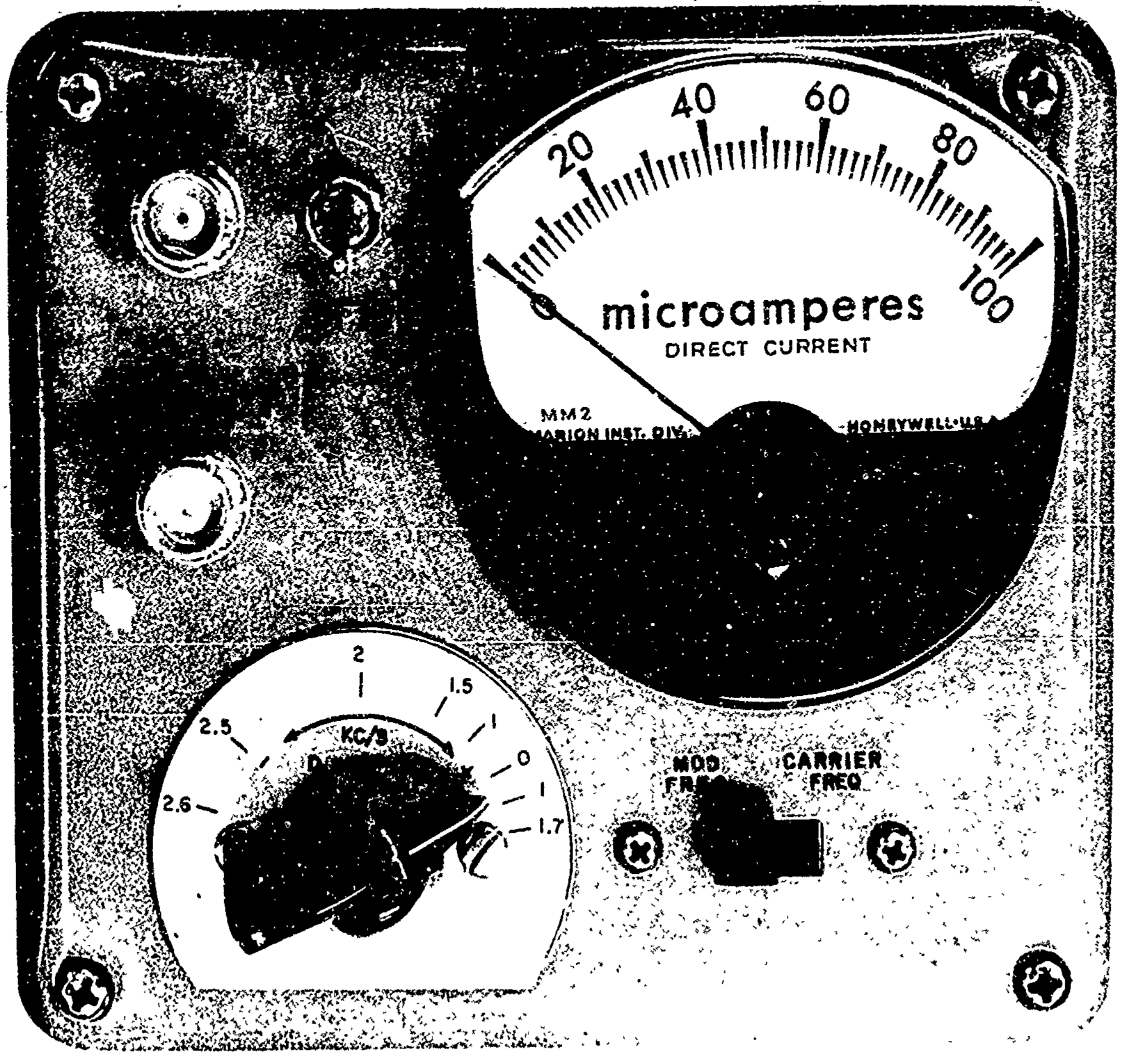
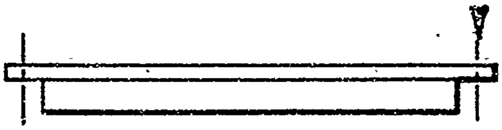


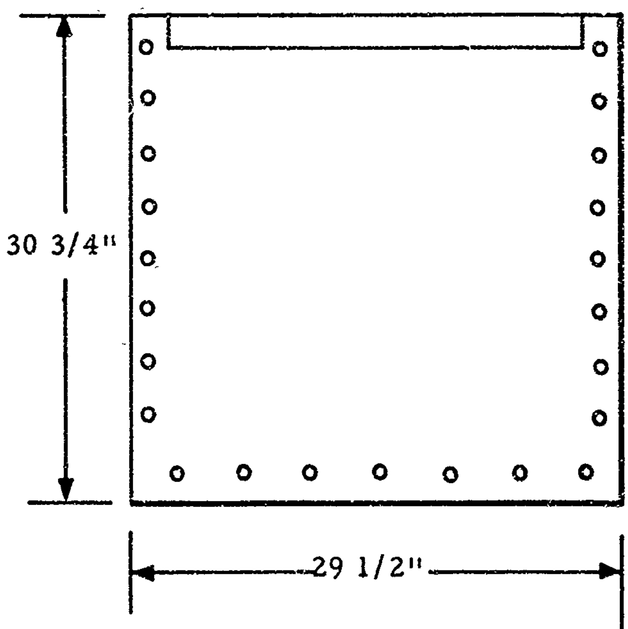
Figure 3-110. Front Panel of Telemetry Test Receiver

TABLE XXIV

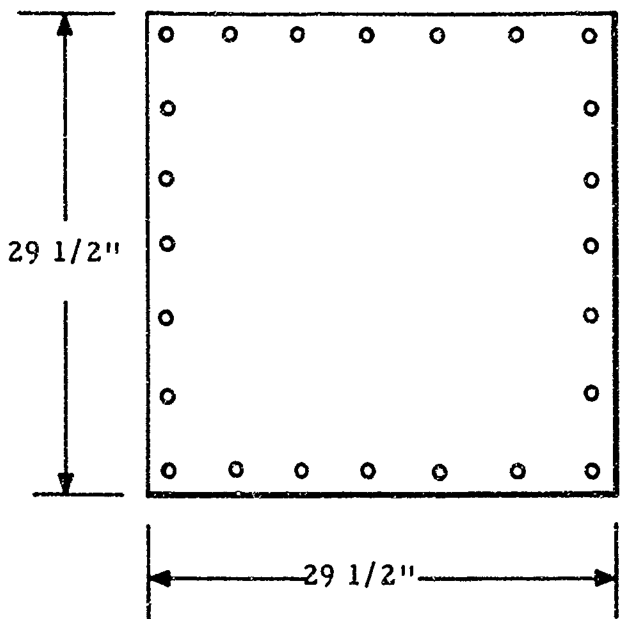
TELEMETRY TEST RECEIVER CONTROL FUNCTION

Control	Function
ANTENNA	Provides connection for antenna
OFF	Energizes receiver circuitry when in alternate position
FM-OUT	Permits monitoring of carrier frequency
KC/S DEVIATION	Tunes local oscillator to provide 10.7 mc if. Indicates deviation (kc) of 136.11 mc telemetry carrier
MICROAMPERES (meter and control)	When switch is in CARRIER FREQ position, meter indicates carrier deviation. When switch is in MOD FREQ, meter indicates modulation deviation (680-780 kc) to 1 cps.

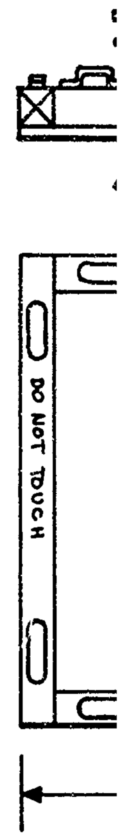




SIDES 2 & 4



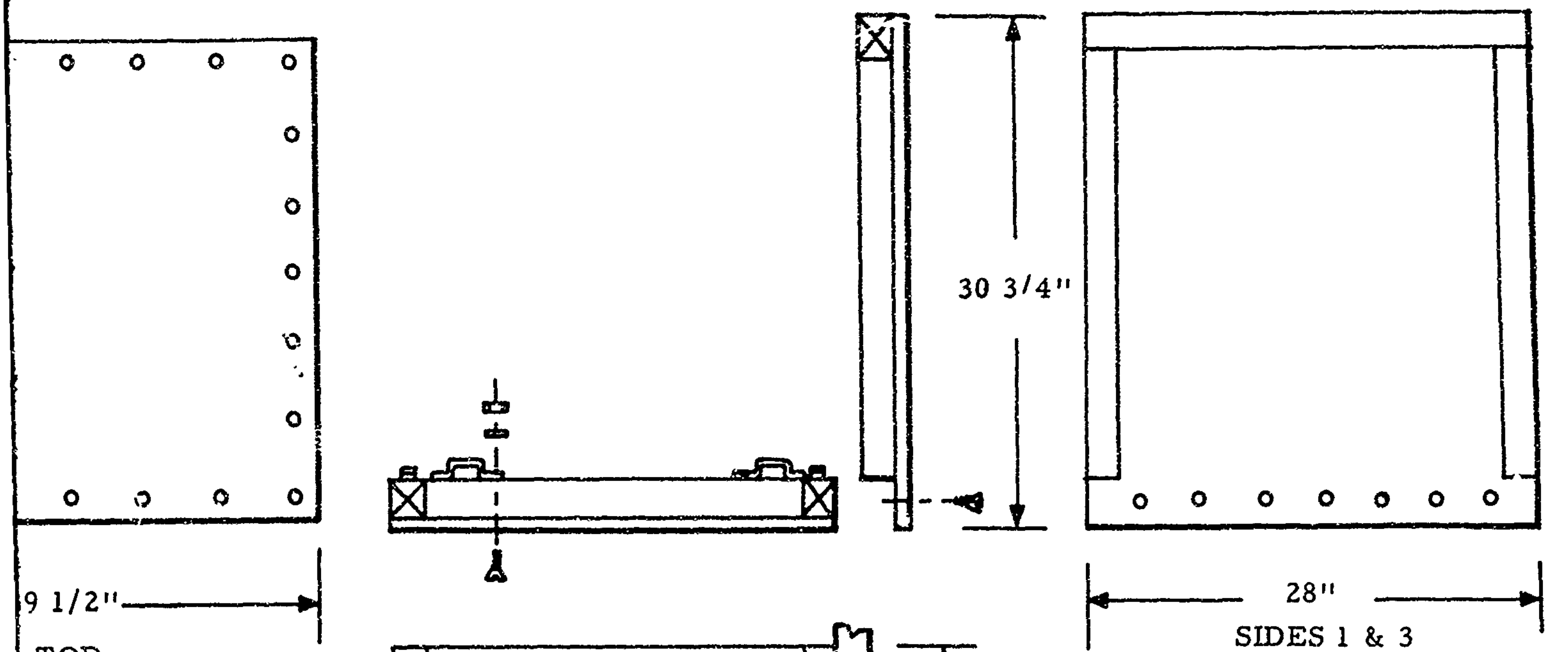
TOP



NOTES

1. UNPAINTED, FH WOODSCREWS USED FOR FASTENING.
2. CORNER-BRACES ARE 2 x 2 CLEAR LUMBER, GLUED IN PLACE.
3. BOX SIDES, TOP, AND BOTTOM ARE 3/4 INCH PLYWOOD.
4. DRAWER PULL HANDLES (MOUNTED ON BOTTOM CORNER-BRACES) ARE FASTENED WITH FLAT HEAD MACHINE SCREWS AND NUTS.
5. "DO NOT TOUCH" WARNINGS ARE STENCILED ON ALL FOUR BOTTOM CORNER-BRACES.

A



LACE.

-BRACES)
NUTS.
R

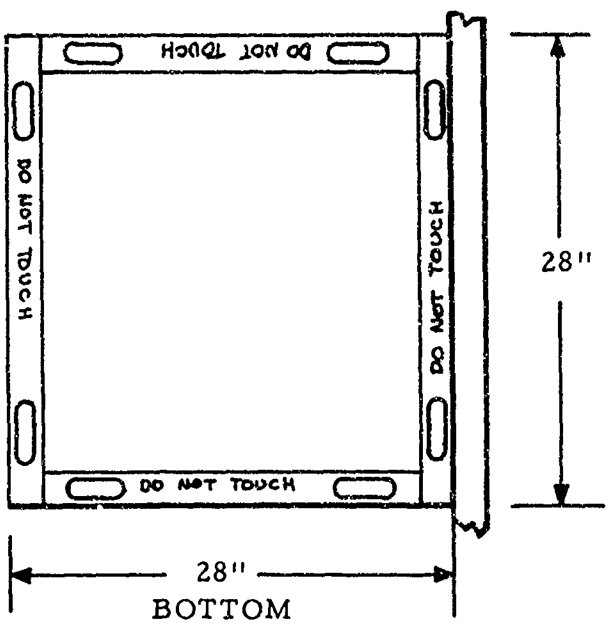
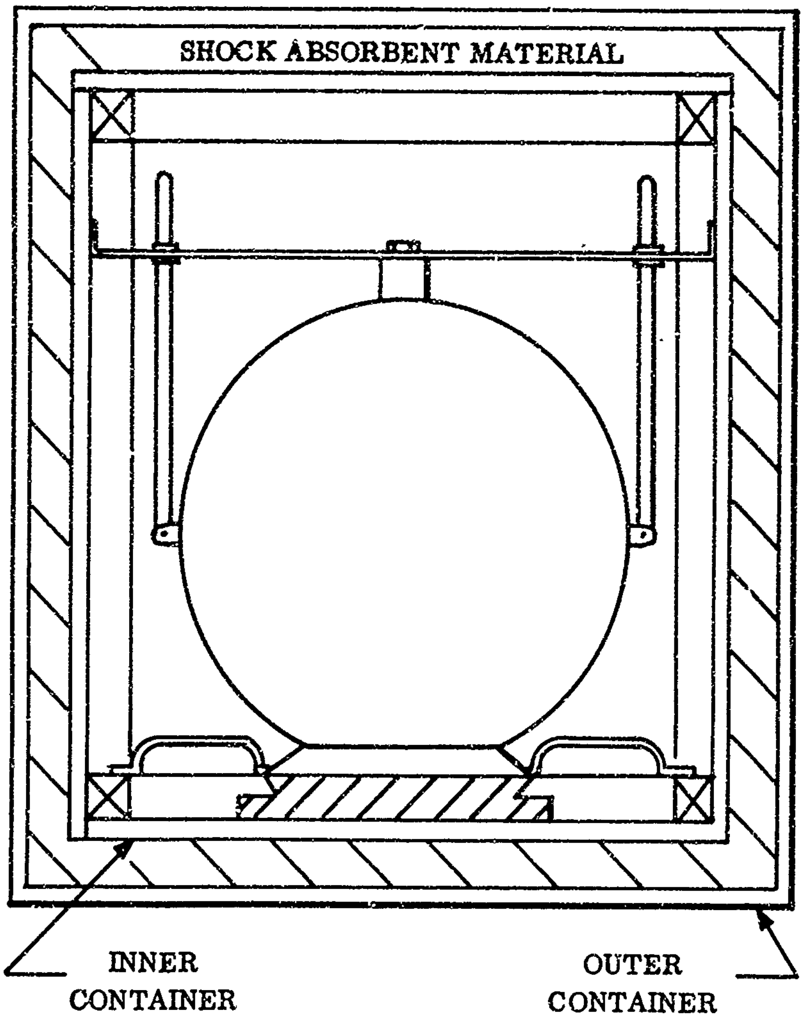


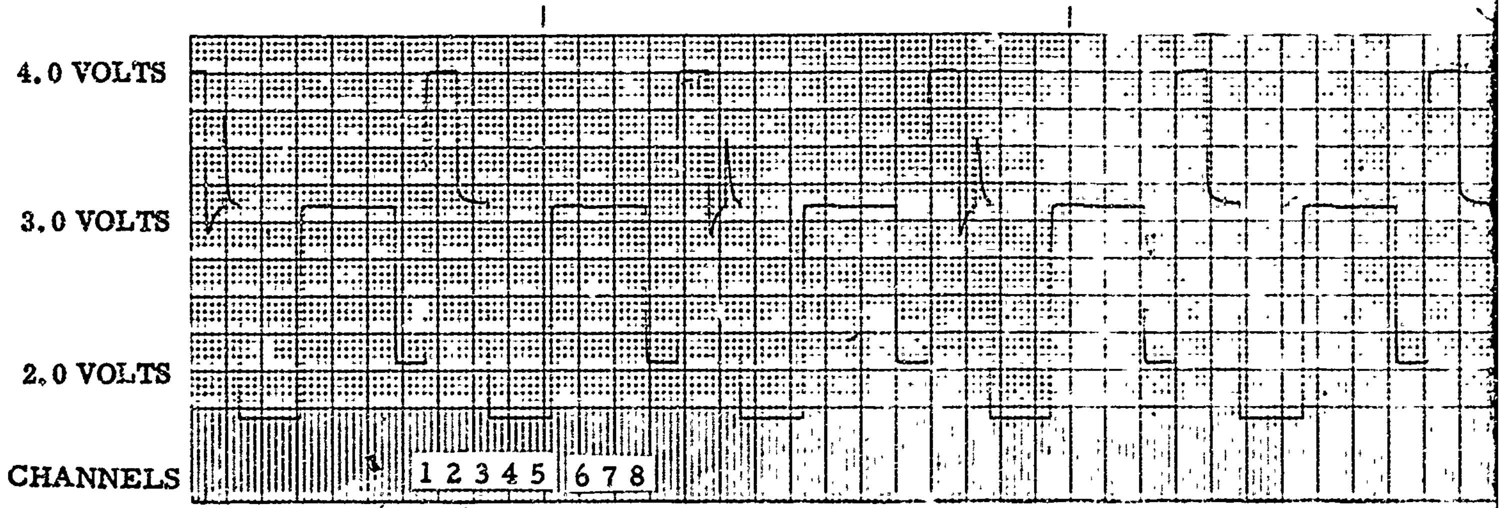
Figure 3-111. Construction Details of Spacecraft Shipping Container

B



NOTE: INSIDE DIMENSIONS OF OUTER CONTAINER SHALL BE 4 INCHES LARGEP THAN OUTSIDE DIMENSIONS OF INNER CONTAINER. 2-INCH THICK BATTS OF RUBBERIZED HORSEHAIR SHALL BE INSERTED AS SHOCK ABSORBERS.

Figure 3-112. Geodetic Spacecraft in Shipping Containers

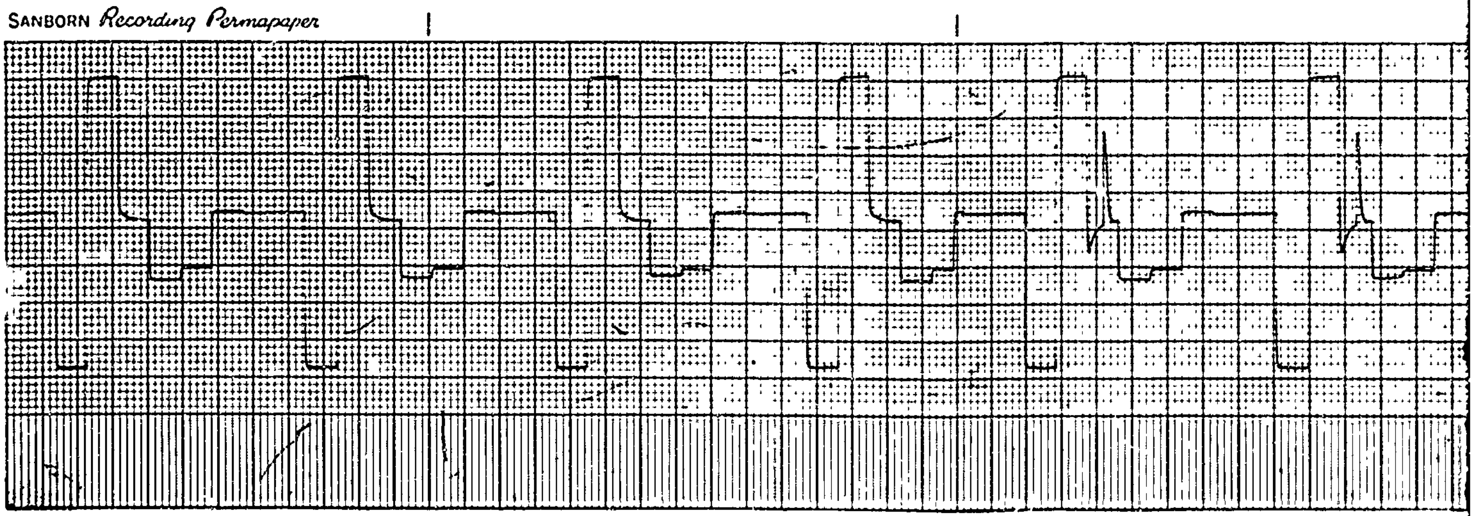


NON-SEPARATION SIGNAL

SEPARATION SIGNAL

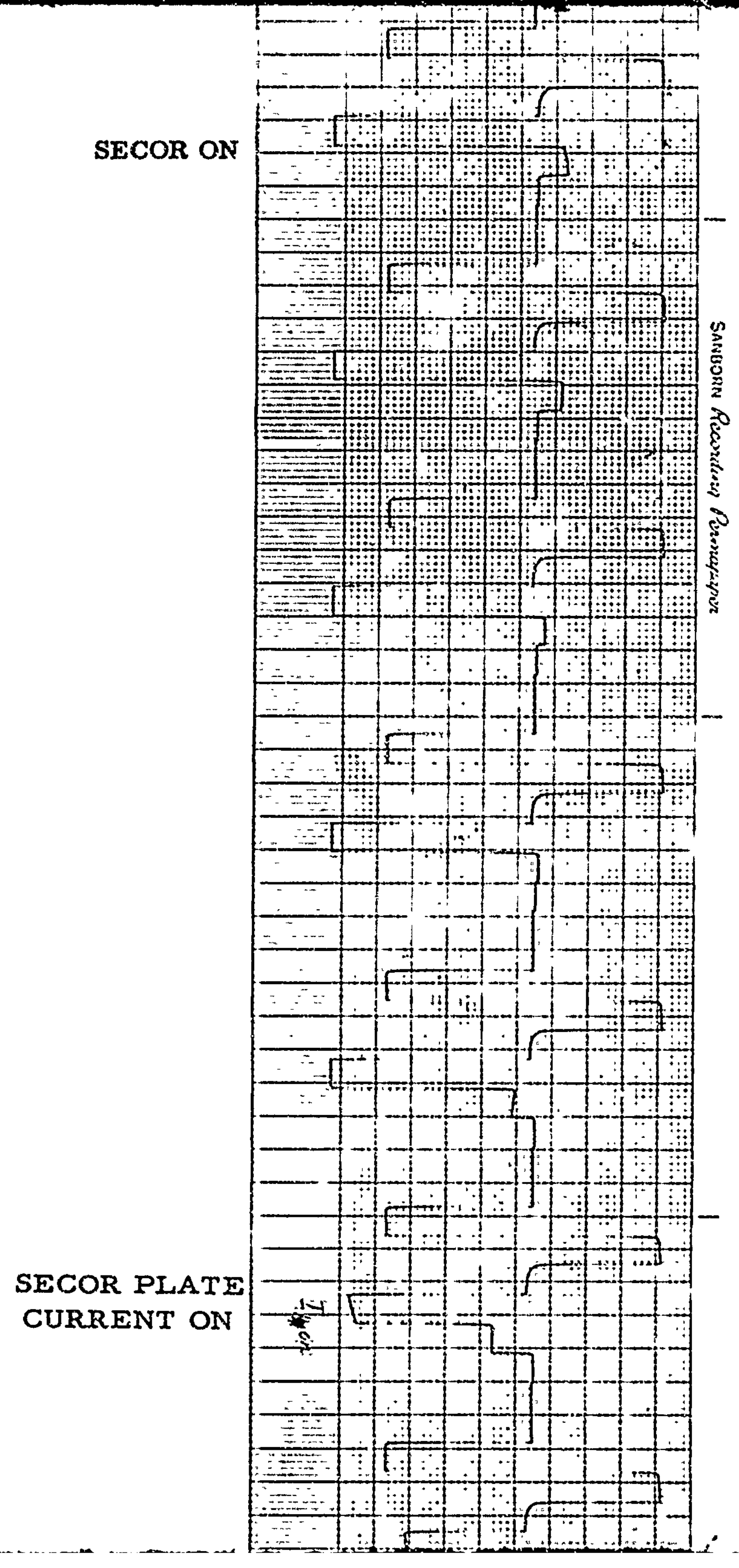
RECEIVER CHECK SIGNAL

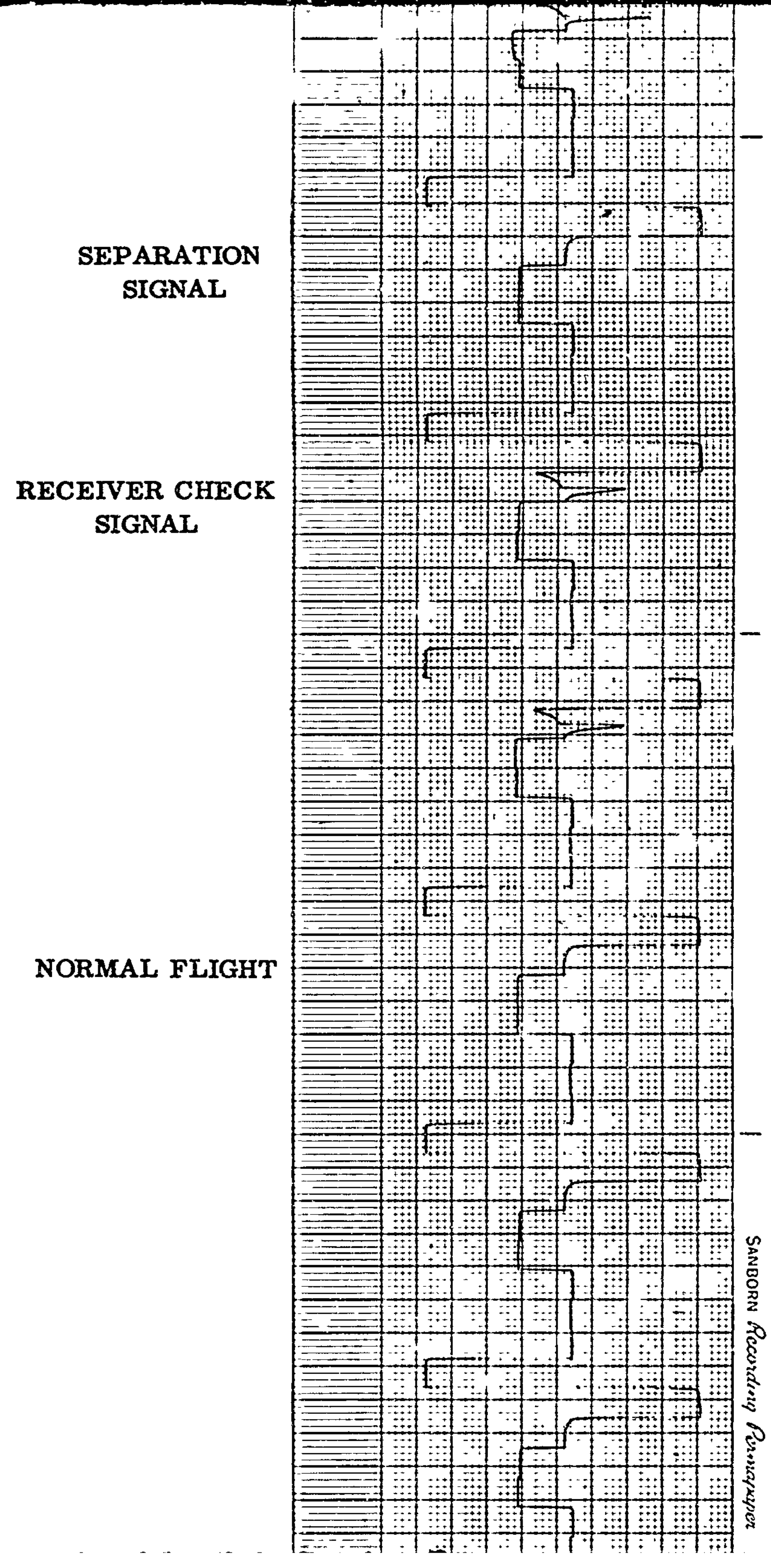
NORMAL



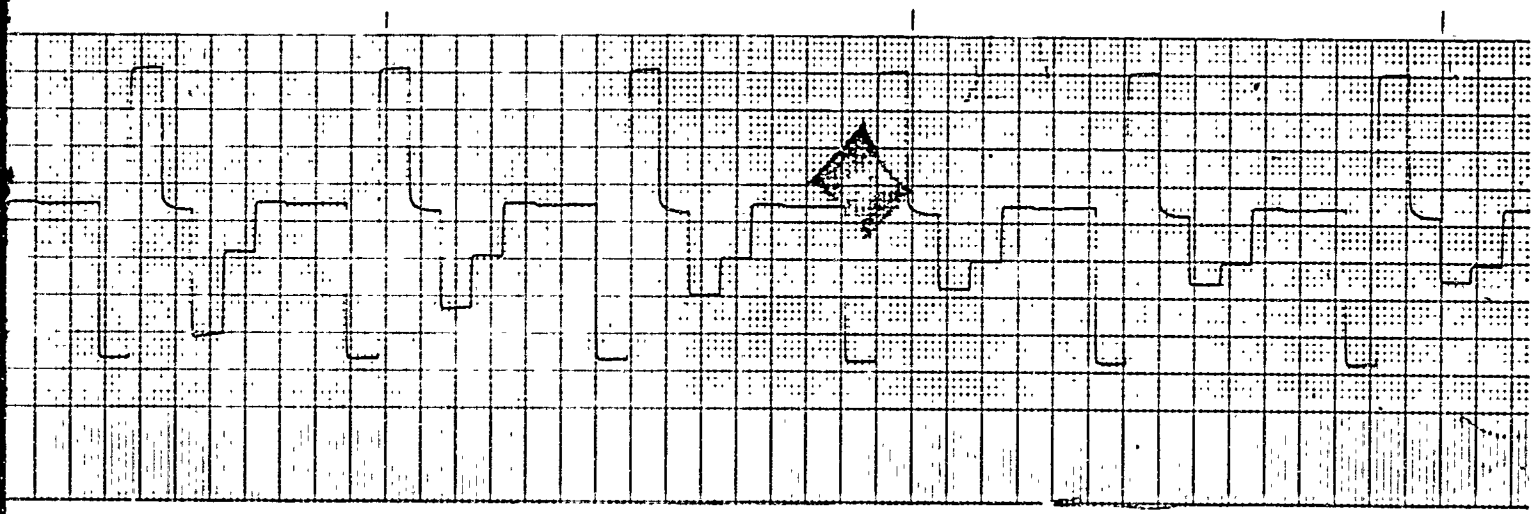
NON-SEPARATION SIGNAL

A


SECOR ON
SANBORN Recording Permapaper
SECOR PLATE CURRENT ON


SEPARATION SIGNAL
RECEIVER CHECK SIGNAL
NORMAL FLIGHT
SANBORN Recording Permapaper

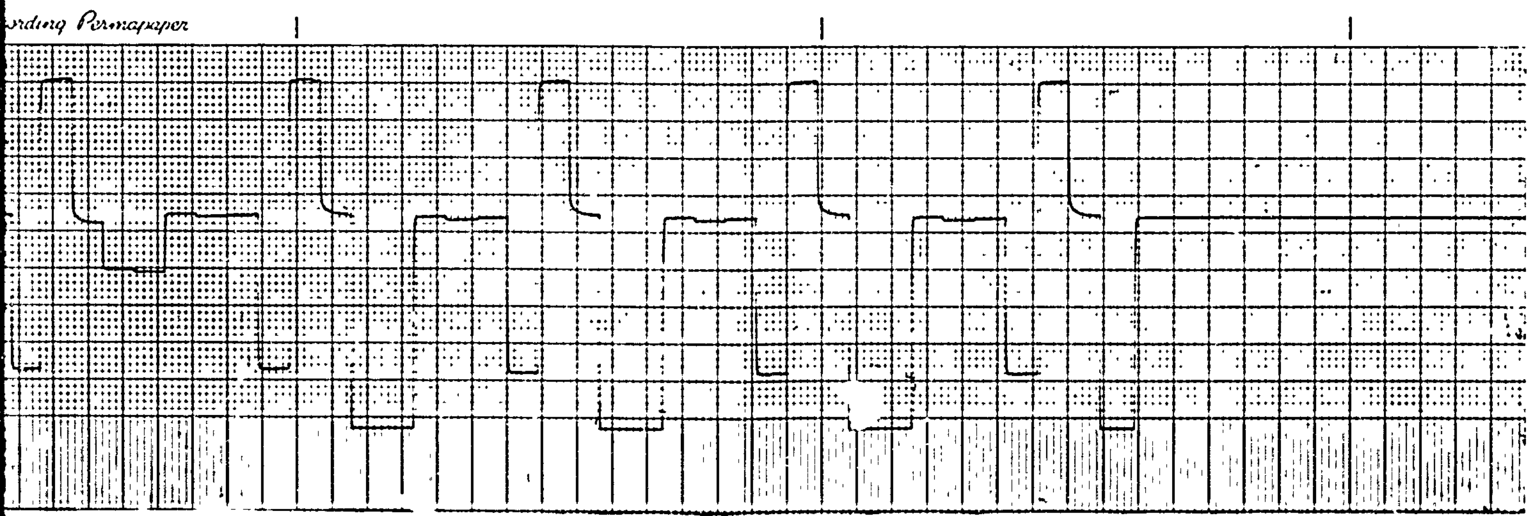
B



11/20/61 GEODETIC SPACECRAFT NO. 3

LINE VOLTS 1201 AMBIENT TEMPERATURE 23.9°C

TM COMMUTATOR OUTPUT TAPE SPEED 5 MM/SEC



SECOR OFF

Figure 3-113. Typical Strip-Chart of Telemetry Commutator Output

C

SECTION IV
DISCUSSION

4.1 Scope of Section. This section contains a discussion of the operational and technical problems encountered in the design, fabrication, assembly, and testing of the major subsystems of the Geodetic Spacecraft. This section presents Cubic's approach to the problem in each instance and includes both successful and unsuccessful investigations.

4.2 Problems Encountered in Development of Mechanical Structure.

4.2.1 The Caleb Vehicle in Geodetic Spacecraft. The Caleb vehicle, described in Exhibit "D" of RFQ 61-338, would have required the development of an entirely new spacecraft of much smaller size than the one provided for the Scout vehicle. The development of a new spacecraft or satellite would have required discarding the engineering, testing, and tooling available for the 20-inch sphere and beginning this effort again, precluding fulfillment of a 3-month delivery schedule and substantially increasing the cost of the few units under procurement. The largest spherical spacecraft containable in the Caleb payload is 15 inches in diameter. Nonspherical payloads can be somewhat larger, but would have required detailed stress analysis if any re-entrant shape were attempted. The smaller sphere would impose limitations on the TR-17 duty cycle. The interior volume would impose severe problems in packaging the TR-17 (whose volume is 174 ~~cw~~ cu in.), any reasonable number of Nicad batteries, the thrust structure, antenna erectors, etc. While not impossible, this effort again precluded fulfillment of a 3-month delivery. For these reasons Cubic recommended that employment of the Caleb launch vehicle could not be favorably considered in providing a Geodetic Spacecraft on the requested 3-months delivery.

4.2.2 Finalization of Separation Fixture Design. As Raymond Engineering Company had manufactured separation fittings for the Vanguard II spacecraft, Brooks and Perkins had agreed to furnish a Raymond separation fitting if it was usable without modification after the type of launch vehicle had been resolved by GIMRADA. As GIMRADA had not yet determined the type of vehicle, Raymond Engineering Company was not able to continue a concurrent fabrication of the separation fixture. However, until GIMRADA resolved the vehicle, Cubic, Brooks and Perkins, and Raymond agreed to conduct a mutual liaison effort in order to determine the type of fitting used on both the proposed vehicles.

4.2.2.1 On 6 March 1961 Brooks and Perkins informed Cubic that Raymond Engineering Company had never fitted a separation mechanism for the final stage of the Scout vehicle, and had no information on the type of fitting which might be required. Brooks and Perkins informed Cubic that the Scout final stage is manufactured by Chance Vought (Astronautics Division), Dallas, Texas, and that they were sending drawings on the separation areas. In the interim, Brooks and Perkins would plan on a Scout configuration and a modified Vanguard II type separation fitting as manufactured by Raymond.

4.2.2.2 After a visit to the NASA Goddard Space Flight Center and Naval Research Laboratory, and a detailed investigation, Cubic learned that the fourth stage of the Scout vehicle rocket motor is the A. B. L. Model 248-AR also used for the Thor-Able-Star and Vangurad II vehicles. Further investigation by Cubic revealed that this rocket motor is equipped with a standard payload mounting flange, that there is no special equipment for the Scout configuration of this motor, and that this conclusion could be verified through NASA.

4.2.2.3 On 14 March, Brooks and Perkins informed Cubic that they necessarily were proceeding with the Raymond separation fixture, and would deliver it as a part of the spacecraft structure. On 16 March GIMRADA ordered Cubic to hold off fabrication of the separation fixture until they had resolved the vehicle type.

4.2.2.4 On 28 March GIMRADA stated that they still had no firm vehicle, but that they had selected arbitrarily, the LOFTI-type mounting ring anticipating a pick-a-back aboard a Transit spacecraft. On 31 March the GIMRADA contracting officer requested that the separation fixture be identical with that used on LOFTI, NRL, and be made compatible with the Transit pick-a-back mount and the modified Scout mounting flange. This fixture was built as an attachment to the first Spacecraft and as an integral part of the other two units.

4.2.2.5 Cubic was unable to procure a clamping device compatible with the mounting flange. The Applied Physics Laboratory was contacted and they furnished drawings on the shipping clamp ring which they built for the Transit payloads. Cubic fabricated similiar clamps for the purpose of handling and shipping the Geodetic Spacecraft.

4.2.3 Modification of Shell to Accommodate Redesigned Solar Cell Plaques. Originally Cubic proposed a plano-spherical plaque whose substrate would be supplied by Brooks and Perkins from the same tooling used in fabricating the satellite shell. The completed plaques would be glued or riveted to the intact shell of the Spacecraft. However, further investigation revealed that a reliable mechanical bonding to the shell could not be achieved. In order to avoid the possibility of resonance buildup, Cubic requested a modification in the shell to accommodate a completely circular-plano plaque.

4.2.3.1 Shell modification required that three 9.065 inch diameter circular openings be cut in each hemisphere in order to accept the new plaques. Reinforcement rings were riveted into each opening, and the plaques were held in place by threaded mating retaining rings. The final mechanical design was excellent, and the shell was structuarally stronger.

4.2.4 Repair of Shell for Spacecraft No. 2. Upon arrival of the shell from Ft. Belvoir where it received the thermal coating, a dent was observed in the lower hemisphere adjacent to the separation fixture. The shell was mated to its internal structure, and the dent carefully tapped out by hand. The assembled Spacecraft was placed on a rotating table, and the surface continuity measured with a dial indicator. (See figure 4-1.) As the surface indicator showed that the surface was within tolerance, Cubic considered the problem solved.

4.2.5 Investigation of Other Spacecraft Configurations. On 4 April, Ft. Belvoir requested that Cubic submit a proposal for two alternate Geodetic Spacecraft which could be utilized for a triple launch on Transit 4B.

4.3 Problems Encountered in Development of Electronics Packages. According to the original purchase description for the Geodetic Spacecraft, Cubic was required to furnish a beacon transmitter as part of the supplementary electronics equipment for each Spacecraft. The beacon transmitter was required to permit NASA tracking facilities to determine the actual orbit of the Spacecraft and to assist the SECOR ground station antennas in acquiring the Spacecraft. The transmitter was to have operated in the 136.11 mc band at an output of 50 to 100 mw.

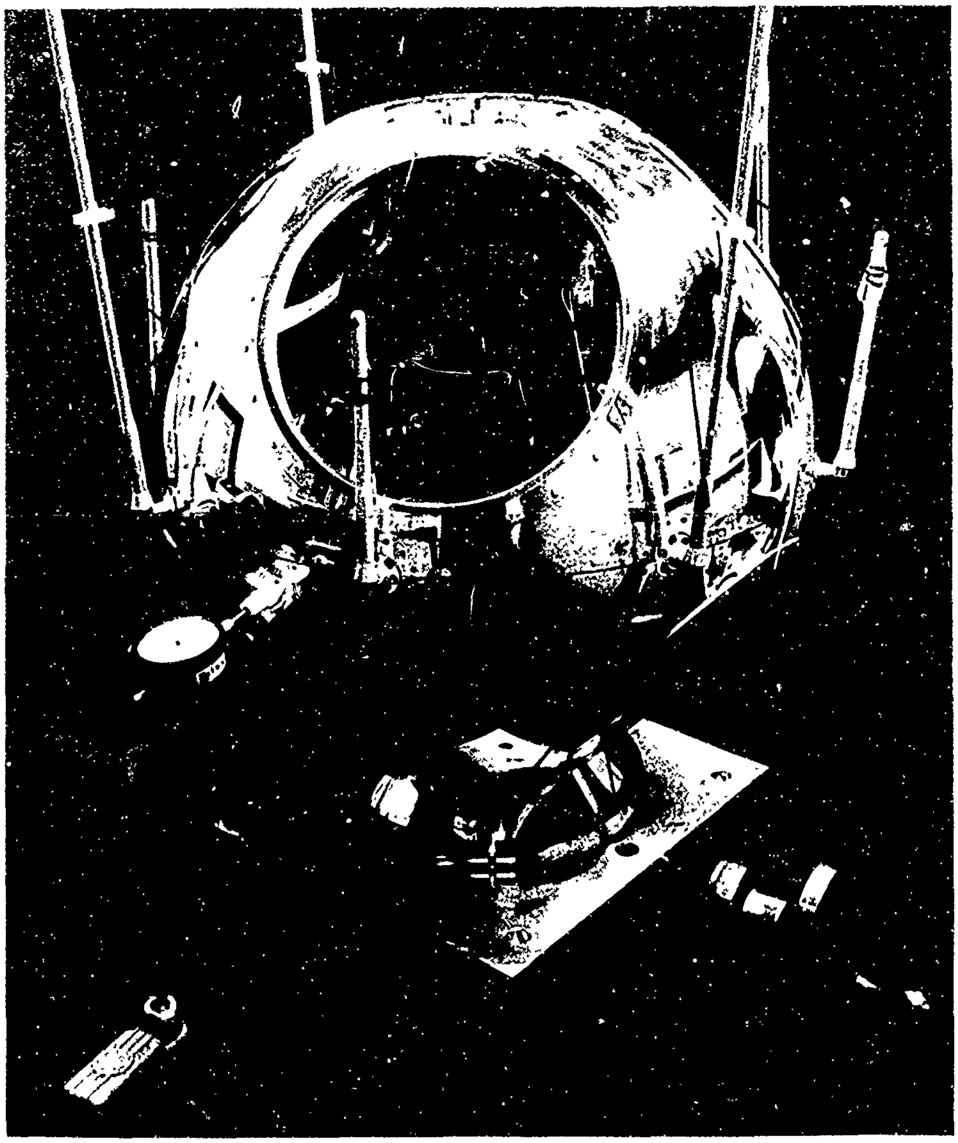


Figure 4-1. Measurement of Spacecraft No. 3 Shell Continuity

4.3.1 After a visit to NASA Goddard Space Flight Center and a detailed study of the requirements, Cubic recommended that a single transmitter be utilized to fulfill both the beacon and the telemetry requirements. It was recommended that this transmitter broadcast continuously (more reliable than switching the two units on and off as required).

NOTE

> Switching would have been necessary as only one frequency, 136.11 mc, had been assigned for both transmitters.

The telemetry transmitter operating on 136.11 mc at 100 mw would provide sufficient power for the NASA tracking functions. The low power may be used becuase of the narrow bandwidth of the telemeter. Similar applications on other projects have proven suitable for NASA Minitrack sites. The transmitter would be turned on at launch and remain on as long as NASA tracking or telemetry is required. An additional command channel in the SECOR transponder would permit the transmitter to be turned on and off from the ground. GIMRADA accepted Cubic's proposal at the recommendation of NASA and NRL, and the requirement for the beacon transmitter was dropped.

4.4 Problems Encountered in Development of Telemetry Subsystem. As the environmental temperature was considered to be mild, elaborate packaging techniques were not considered necessary to protect the transistor circuits. This permitted the use of germanium transistors. The use of silicon elements would have required greater power, resulting in shorter operating time for the SECOR transponder. In addition, shielding would have been required, resulting in greater weight.

4.4.1 During the calibration procedure of the prototype telemetry boards, the voltage regulation at -20°C was poor and beyond allowable tolerances. This condition was remedied by changing the 3- and 6-volt regulators to draw more current in the Zener diodes.

4.4.2 Because of intermittent difficulty with one of the telemetry printed circuits boards after vibration and thermal shock (-10°C), it was decided to enlarge the contact area around the eyelets.

The existing boards were modified and new printed circuit boards were fabricated for the Spacecraft. In the redesigned boards there was a possibility of the card containing the 3-volt regulator shorting out against the frame because of vibration. The frame beneath the card was insulated to prevent this.

4.4.3 Originally an 8 sec frame was planned, however circuit components resulted in the present 6.64 ±0.20 sec frame. As the parameters change only at slow rates a faster system was not required and the circuit parameters were satisfactory.

4.5 Elimination of the 136.11 MC South Pole Antenna. In the original design Cubic planned to utilize north and south pole antennas for the telemetry subsystem. However, antenna radiation patterns performed by Temec, Inc. indicated that the south pole dipole added very little to the total radiation coverage. Also, the spiral design was complicated and unproven. These considerations plus the possibility that improper separation from the launch vehicle could short out the antenna, resulted in the decision to eliminate the south pole antenna.

4.6 Problems Encountered in Development of Power Supply Subsystem.

4.6.1 Battery Package Pressurization. As an adequate pressure seal was not available, Cubic resolved to pot the batteries and leave the power converter exposed to ambient pressure. A corona condition experienced at approximately 80,000 ft could conceivably produce arcing in the converter. During its year in predicted orbit, had the seal leaked, the converter would have been exposed to an equivalent corona atmosphere for an extensive length of time, effectively limiting the usability of the transponder. At present the transponder will only be unusable during the vehicle's brief pass through the corona belt shortly after launch. The SECOR system does not require operation during ascent; therefore the limitation is academic.

4.6.2 In the selection of a suitable potting compond, the first one did not withstand the pressure test, and the Dow Corning 502 had a specific gravity greater than 1. The Eccosil 4640, with a specific gravity of 0.65 was finally selected and withstood all pressure tests.

The tolerance of the package is small enough to insure a tight fit in the tube, with no movement.

4.7 Cleaning of Thermal Coating and Lubrication of Shell Fittings. Cubic utilized a commercial brand of ethyl alcohol for purposes of cleaning the SiO coated shell, and for lubricating the solar cell plaque retaining rings.

SECTION V
CONCLUSIONS

5.1 General. This section contains statements of the logical conclusions resulting from an analysis of the data derived during the development of the Geodetic Spacecraft.

5.2 Mechanical Structure.

(1) As the mechanical structure of the Spacecraft is much more rugged than originally anticipated, it need not be machined to the high specified tolerances. Other than the degree of finish on the outer shell required fo. reflectivity, the mechanical aspects of launch, not those of outer space, are the controlling factors.

(2) In the present configuration, the internal thrust structure was necessary to provide adequate mechanical support for the antennas and the thin outer shell. If a thicker shell were fabricated from 0.050 inch aluminum, the internal bracing could be eliminated. The instrumentation support tube would be a continuation of the mounting plate, and would be extended to provide support for the shell at the north pole. The separation band around the equator of the shell would support the antennas.

(3) The large number of small screws utilized to assemble the Spacecraft can be replaced by fewer, larger screws. This would facilitate assembly and disassembly of the Spacecraft and would not affect the mechanical strength or performance.

(4) The solar cell plaque retaining rings have proved to be an excellent design. They are mechanically strong and are self-locking and will not loosen during vibration. The rings enable rapid removal of any of the plaques, and thus provide quick access to the spacecraft interior without requiring disassembly of the shells. When installed, the rings do not permit the plaques to slip or move, even if the plaques are undersize.

5.3 Electronics Subsystems.

(1) The SECOR receiver can also be utilized as a command receiver without affecting the performance of the transponder function.

(2) Because of the slow rate of change in the telemetry data, a commutation rate much slower than the original one second per channel could be utilized without loss of information.

5.4 Solar Cell Plaques.

(1) The major problem in the fabrication of the solar cell plaques is obtaining the honeycombed panels. A ready supply of 12 per cent solar cell is available from several sources and, if the plates are available, the plaques can be quickly assembled.

5.5 Battery Package.

(1) The performance of the Sonotone type WS-103 Nicad batteries has been excellent. However, there is some question regarding the requirement for encapsulating the battery package. If a pressure seal is required, it can most likely be accomplished by simpler means. If batteries are potted in future Spacecraft they should be arranged in a flatter configuration. This design would be more easily removed from the Spacecraft than the present long cylindrical configuration.

5.6 Antennas.

(1) An analysis of the antenna pattern shows that the optimum performance of the ground stations can be achieved by using a combination of horizontal and vertical polarization. Physical properties of the Spacecraft (size, areas covered by solar cells, requirement for folding antennas, etc.) place a limitation on the patterns of coverage that can be achieved by the Spacecraft antennas. It will require less time and effort to change the ground antenna feed system than to improve the spacecraft antenna coverage.

5.7 Thermal Stabilization.

(1) A simple cover over the telemetry printed circuit boards will reduce the thermal excursions of the boards by approximately $10^{\circ}C$.

(2) A thermal coating of either SiO or black and white paint can be utilized to provide the thermal stabilization required by the Spacecraft.

SiO is preferred because its application can be more closely controlled, and because there is some question regarding the properties of the paint after prolonged exposure to outer space conditions. The SiO coating should be applied near the end of the program rather than at the beginning. This would eliminate the precautions and inconvenience in handling the coated shells during the development phase of the program.

5.8 Over-all Development of Program.

(1) Most of the problems encountered in the mechanical design of the Spacecraft resulted from the lack of knowledge regarding the type of launch vehicle. The final external configuration of the Spacecraft remained in doubt throughout most of the program. Fortunately, this problem was adequately planned for, and a minimum of redesign was required when the launch vehicle was selected and the external configuration finalized. In future programs the vehicle should be selected prior to commencement of the spacecraft development in order that the Spacecraft may be tailored to a specific configuration rather than to a range of configurations.

(2) The customer should approve all detail designs and environmental test programs before their enactment, in order to ensure agreement and fulfillment of design objectives.

(3) A customer representative should be present at each major test to approve any required deviation from the program due to test setup limitations.

SECTION VI
RECOMMENDATIONS

6.1 General Recommendations. With regard to spacecraft development, it is recommended that the following areas receive further study and/or evaluation.

6.2 Spacecraft Configuration for Caleb Vehicle. The present 20-inch Geodetic Spacecraft has a low density factor with regard to the electronic packaging. All of the electronics used in the present Spacecraft could be repackaged in a smaller external configuration suitable for launch by the NRL Caleb vehicle. It may be desirable to depart from the use of a spherical shell in order to better utilize the space available. However, a nonspherical shell will not affect the performance of the Spacecraft or of the SECOR transponder.

6.2.1 The 25-pound weight limit required for the Caleb could be achieved by eliminating the second set of batteries presently aboard the Geodetic Spacecraft. This would shorten the time per day that the transponder could be operated but would provide a Spacecraft suitable for use by the Geodetic SECOR.

6.2.2 Because of the relative low cost of injecting a spacecraft into orbit by the Caleb, and the requirement for a large number of Geodetic Spacecraft for the operational system, it is recommended that a feasibility study be initiated immediately. The study must include investigation into the spacecraft design problems, and also into the use and availability of the launch vehicles and the operational concepts. It is recommended that the investigation be initiated as soon as possible in order that the new design problems may be resolved in time for an orderly integration into the program.

6.3 Agena as a Launch Vehicle. During the investigation of possible launch vehicles it was learned that numerous launch possibilities are available aboard an Agena rocket or as pick-a-back on a Discoverer. Each Discoverer has the capability of carrying several 15-inch diameter spheres into orbit. Because of the short life (30 days) of a spacecraft launched into the relatively low Discoverer orbit, the use of solar cells to recharge the batteries would not be required. A very simple Geodetic Spacecraft consisting primarily of

of a battery package, a SECOR transponder and the required antenna system, could be built and designed for ejection from the Discoverer while in orbit. The advantages of this technique as compared to having the transponder flown as a part of the Discoverer are numerous. First, it eliminates the interference problem which would impose a serious limitation on the utility of the transponder. Second, it would overcome the loss of tracking data caused by the tumbling of the large Discoverer vehicle. Third, it would permit operation of the SECOR independent of the operation of the Discoverer payload.

6.3.1 It is therefore recommended that an investigation be initiated into the design of a Geodetic Spacecraft which would be compatible for launch by the Discoverer spacecraft and other programs using the Agena vehicle.

6.4 Use of Geodetic Spacecraft for Additional Experiments. It is recommended that a study be initiated into the possibilities of utilizing the present spacecraft configuration for additional space experiments. There is considerable unused space in the present spacecraft design. Also, adequate power and telemetry channels are available to permit experiments other than the Geodetic SECOR. If the Spacecraft were expanded to a 24-inch diameter, there would be sufficient volume and solar cell area available to allow numerous simultaneous experiments during the SECOR mission. Possible experiments include the investigation of ionspheric effects on rf transmissions, the investigation of physical force stabilization techniques, the use of Spacecraft to relay tracking data to central computing centers for real-time computation of the station coordinates, and the employment of combinations of satellite-borne TV cameras to correlate the SECOR survey data. In each of these programs, Cubic would integrate the existing hardware into an operational spacecraft system.

6.5 Use of Geodetic Spacecraft to Verify SECOR Range Measurements. A program is recommended to investigate the types of lamps (and their installation requirements) suitable for use in a Geodetic Spacecraft for purposes of tracking by ballistic cameras. As soon as the Geodetic SECOR System becomes operational, there will be a requirement for a calibration satellite. This Spacecraft will carry a flashing light in addition to the normal SECOR equipment.

Ballistic cameras positioned around the SECOR ground stations will obtain tracking data from the flashing light while the SECOR is tracking the satellite. This will provide a check on the accuracy of the basic range measurements of each of the SECOR stations.

6.6 Multiple Target Tracking. Cubic proposes a study into the development of a multiple-tracking SECOR system utilizing three or more Spacecraft. This longer term program would involve a modification of the ground stations for multiple-target tracking similar to the SECOR at WSMR. This would permit three Spacecraft to be tracked simultaneously, thereby establishing a baseline in space. When referenced to this baseline, other satellites performing photogrammetry beyond the radio horizon could be provided geodetic coordinates.

6.7 Alternate Antenna Subsystems. It is recommended that a study be conducted into the use of alternate antennas and antenna arrangements which may improve the coverage particularly at the higher frequencies. Because of the phasing problems of the array on the small sphere, it is not possible to predict accurately the final antenna patterns. Radiation patterns must be obtained for alternate type antennas, such as V's in an attempt to fill in some of the nulls exhibited in the simple dipole arrangement. The initial requirements for folding the antennas into the nose faring of the Scout precluded the use of such antennas initially.

6.8 Additional Recommendations. In addition, Cubic recommends studies of the possibility of designing a miniature SECOR transponder which could incorporate the function of a telemeter for use in the smaller proposed Spacecraft.

APPENDIX A

GEODETIC SPACECRAFT NO. 1
TELEMETRY DATA

Table A-1
Geodetic Spacecraft No. 1

TIME	CHAMBER THERMAL CONDITION	CHAMBER PRESSURE mmHg	BATT SUPPLY VOLTS	SELECT CALL OK	TM "ON" OK	TM "OFF" OK	CURRENT			VOLTAGE	
							SECOR RCVR	TM	SECOR XPNDER+TM	SOLAR CELL	BATT NO
10:05	Room	30.00	12.33		x	x	28 ma	44 ma			12.49
10:14	Room	30.00	11.83	x			On Cont.	On Cont.	1.85A	0.72	12.49
10:19	Room	30.00	12.49	x							
17:00			11.89				28 ma	44 ma		0	12.49
17:10		$2x10^{-5}$	11.89	x			28 ma	44 ma	2.0 A		
17:30	Full heat	$3.7x10^{-5}$	11.85		x	x	28 ma	44 ma		5.19	
17:45	Full heat	$5x10^{-3}$	11.89	x	x	x	28 ma	44 ma		6.19	12.49
18:00	Full heat	$6x10^{-2}$	11.89				28 ma	44 ma		.29	12.49
18:15	Liquid N	$2.5x10^{-6}$	11.89							4.49	12.49
18:16	Heat off										
18:30	Liquid N	$2.3x10^{-5}$	11.89	x	x	x	28 ma	44 ma		0	12.49
18:45	Liquid N	$2x10^{-5}$	11.89	x	x	x	28 ma	44 ma		0	12.49
18:50	Liquid N	$2x10^{-5}$	11.39	x	x	x	28 ma	44 ma	1.85A	0	12.49
19:00	Liquid N	$2x10^{-5}$	11.89	x	x	x	28 ma	44 ma		0	12.49
19:15	Liquid N	$2x10^{-5}$	11.89	x	x	x	28 ma	44 ma		0	12.49
19:30	Liquid N	$2x10^{-5}$	11.89	x	x	x	28 ma	44 ma		0	12.49
19:47	Liquid N	$2x10^{-5}$	11.89	x	x	x	28 ma	44 ma		0	12.49
19:50	Max.heat										
20:00	Max.heat	$2x10^{-5}$	11.89				28 ma	44 ma		7.17	12.49
20:15	Max.heat	$3x10^{-5}$	11.89	x	x	x	28 ma	44 ma		6.11	12.49
20:17	Max.heat	$3x10^{-5}$	11.41	x					1.85A	6.23	12.75
20:30	Max.heat	$3x10^{-5}$	11.87	x	x	x	28 ma	44 ma		5.82	12.49
20:45	Heat	$4x10^{-5}$	11.87	x	x	x	28 ma	44 ma		.72	12.49
21:00	Cold	$2x10^{-5}$	11.87	x	x	x	28 ma	44 ma		0	12.49
21:15	Cold	$1.8x10^{-5}$	11.86	x	x	x	28 ma	44 ma		0	12.49
21:20	Heat	$1.8x10^{-5}$	11.41	x	x	x			1.85A	6.27	12.49
21:30	Heat	$1.8x10^{-5}$	11.86	x	x	x	28 ma	44 ma		5.79	12.49
21:45	Heat	$2.3x10^{-5}$	11.84	x	x	x	28 ma	44 ma		5.60	12.49
22:00	Heat	$2.6x10^{-5}$	11.83	x	x	x	28 ma	44 ma		1.10	12.49
22:15	Heat	$6x10^{-5}$	11.88	x	x	x	28 ma	44 ma		.40	12.49
22:30	Cold	$1.6x10^{-5}$	11.80	x	x	x	28 ma	44 ma		0	12.49
22:45	Cold	$1.8x10^{-5}$	11.79	x	x	x	28 ma	44 ma		0	12.49
23:00	Cold	$1.8x10^{-5}$	11.79	x	x	x	28 ma	44 ma		0	12.49
23:15	Heat	$2.2x10^{-5}$	11.79	x	x	x	28 ma	44 ma		5.60	12.49
23:20	Heat	$2.2x10^{-5}$	11.34	x	x	x			1.85A	5.80	12.72

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e A-1

No. 1 Telemetry Data

LTAGE		TELEMETRY COMMUTATOR OUTPUT								NOTES
		LOW CALIB.	HIGH CALIB.	BATT. E_b	E_k	E_p	SKIN INSIDE	BATT.	SOLAR CELL	CENTIGRADE TEMPERATURES REFER TO THERMISTORS
BATT NO	BATT NO 2	NO 1	NO. 2	NO. 3	NO. 4	NO. 5	NO 6	NO 7	NO 8	
.49	12.29	1.99	4.25	2.66	1.29	1.29	3.25	3.44	3.40	
.49	12.29	2.24	4.50	2.61	2.77	3.39	3.49	3.69	3.60	Door open
		2.31	4.59	2.99	3.50	3.79	3.59	3.73	3.70	
.49	12.29	1.99	4.29	2.49	1.29	1.29	3.23	3.24	3.43	
		2.29	4.59	2.61	3.11	3.31	3.52	3.50	3.71	
		1.99	4.29	2.49	1.29	1.29	3.09	3.19	3.01	3900 Watts
.49	12.29	1.99	4.29	2.49	1.29	1.29	2.69	3.19	2.49	#6-34°, #8-43°
.49	12.29	2.01	4.29	2.49	1.29	1.29	2.39	3.19	2.39	#6-39.5°, #8-43°
.49	12.29	2.02	4.29	2.49	1.29	1.29	2.39	3.09	2.39	Liquid N, heat on
.49	12.29	2.03	4.29	2.49	1.29	1.29	2.69	3.09	2.91	#6-34°, #8-31°
.49	12.29	2.00	4.29	2.49	1.29	1.29	2.99	2.99	3.29	#6-27°, #8-22°
.49	12.29	2.29	4.59	2.29	2.89	2.89	3.29	3.20	3.59	Xpndr "ON"
.49	12.29	1.99	4.29	2.49	1.29	1.29	3.29	3.00	3.59	#6-21°, #8-18°
.49	12.29	1.99	4.29	2.49	1.29	1.29	3.40	3.01	3.69	#6-16°, #8-14°
.49	12.29	1.99	4.29	2.51	1.30	1.31	3.61	3.07	3.86	#6-12.5°,#8-10.5°
.49	12.29	1.99	4.29	2.51	1.30	1.31	3.76	3.13	3.99	#6-10°, #8-6.5°
										4552 Watts
.49	12.29	1.99	4.27	2.51	1.30	1.30	3.61	3.19	3.54	#6-12°,#8-18.5°
.49	12.29	1.99	4.29	2.50	1.30	1.30	3.22	3.22	3.04	#6-23.5°,#8-28.5°
.75	12.54	2.25	4.54	2.37	2.90	2.99	3.35	3.42	3.16	Xpndr "ON"
.49	12.29	2.02	4.30	2.49	1.30	1.31	2.74	3.19	2.54	#6-36°, #8-39°
.49	12.29	2.03	4.32	2.49	1.31	1.31	2.49	3.15	2.44	#6-38°,#8-42.5°
.49	12.29	2.04	4.31	2.49	1.30	1.31	2.72	3.11	2.96	#6-33°, #8-31°
.49	12.29	2.03	4.31	2.49	1.30	1.31	3.01	3.08	3.29	#6-26°, #8-22°
.49	12.29	2.27	4.57	2.35	2.98	2.95	3.26	3.29	3.46	Xpndr "ON"
.49	11.27	2.03	4.31	2.49	1.30	1.31	2.84	3.05	2.79	#6-28°, #8-34°
.49	11.27	2.03	4.32	2.46	1.31	1.31	2.62	3.02	2.53	#6-36°, #8-40°
.49	11.27	2.04	4.33	2.46	1.31	1.31	2.40	2.99	2.40	#6-37.5°,#8-42.5°
.49	11.27	2.05	4.33	2.44	1.30	1.30	2.31	2.92	2.39	#6-40.5°,#8-44°
.49	12.25	2.05	4.32	2.43	1.30	1.30	2.52	2.89	2.80	#6-35°, #8-34°
.49	12.23	2.04	4.32	2.44	1.30	1.30	2.85	2.86	3.15	#6-29°, #8-26°
.49	12.23	2.02	4.31	2.44	1.30	1.30	3.10	2.89	3.39	#6-24°, #8-20°
.49	12.23	2.04	4.30	2.43	1.30	1.30	2.92	2.90	2.90	#6-26°,#8-30.3°
.72	12.51	2.29	4.59	2.30	2.85	2.85	3.10	3.11	3.03	

Table A-1
Geodetic Spacecraft No. 1

NC = Not Checked

TIME	CHAMBER THERMAL CONDITION	CHAMBER PRESSURE mmHg	BATT SUPPLY VOLTS	SELECT CALL OK	TM "ON" OK	TM "OFF" OK	CURRENT SECOR RCVR	CURRENT TM	CURRENT SECOR XPNDER+TM	VOLTAGE SOLAR CELL	VOLTAGE BATT NO 1
23:30	Heat	$2.7x10^{-5}$	11.79	x	x	x	28 ma	44 ma		1.84	12.49
23:45	Heat	$2.3x10^{-5}$	11.79	x	x	x	28 ma	44 ma		.4	12.49
24:00	Heat	$2.4x10^{-5}$	11.79	x	x	x	28 ma	44 ma		.5	12.49
00:15	Cold	$1.8x10^{-5}$	11.79	x	x	x	28 ma	44 ma		0	12.49
00:30	Cold	$1.6x10^{-5}$	11.79	x	x	x	28 ma	44 ma		0	12.49
10:05	Cold	$2x10^{-5}$	11.75	x	x	x	27.5 ma	43 ma	1.70A	0.01	12.50
10:25	Heat	$2x10^{-5}$	11.67	x	x	x	28 ma	43 ma	1.80A	4.83	12.52
10:40	1200W	$2.2x10^{-5}$	11.66	x	x	x	27 ma	44 ma	1.80A	5.40	12.52
10:55	720W	$2.3x10^{-5}$	11.87	x	x		27 ma	43 ma	1.80A	1.59	12.53
11:20	720W	$7x10^{-5}$	11.85	x	x	x	27 ma	44 ma	1.80A	1.52	12.51
11:35	720W	$3x10^{-5}$	11.82	x	x	x	28 ma	44 ma	1.90A	1.44	12.51
11:50	720W	$3x10^{-5}$	11.82	NC	x	x	28 ma	45 ma	1.90A	1.10	12.51
12:10			11.87	x	x	x	28 ma	45 ma	NC	0.02	12.50
12:12	Go cold	$1.9x10^{-5}$	11.87	NC	NC	NC	NC	NC	1.90A		12.80
12:30	Cold	$2x10^{-5}$	12.00	x	x	x	28 ma	47 ma	2.00A	0.27	12.80
12:35	Cold	$2x10^{-5}$	11.97	NC	x	x	29 ma	46 ma	NC	0.01	12.49
12:50	Cold	$3x10^{-5}$	11.96	NC	x	x	29 ma	46 ma	NC	0.02	12.49
12:52	Cold	$3x10^{-5}$	11.96	x	x	x	NC	45 ma	2.00A	0.24	12.79
13:10	Cold	$1.8x10^{-5}$	12.00	x	x	x	NC	44 ma	2.00A	0.21	12.78
13:15	Cold	$1.8x10^{-5}$	12.00	NC	x	x	29 ma	45 ma	NC	0.01	12.44
13:30	Cold	$2x10^{-5}$	12.00	NC	x	x	29 ma	45 ma	NC	0.06	12.49
13:35	Cold	$2x10^{-5}$	12.00	x	x	x	NC	45 ma	2.00A	0.20	12.78
13:40	Cold	$2x10^{-5}$	12.00	NC	NC	NC	NC	NC	NC	NC	NC
13:50	Heat	$2x10^{-5}$	12.00	x	x	x	NC	45 ma	2.00A	2.80	12.79
13:55	1100W	$2x10^{-5}$	12.00	NC	x	x	29 ma	45 ma	NC	2.54	12.49
14:10	1100W	$2x10^{-5}$	12.00	NC	x	x	29 ma	45 ma	NC	2.02	12.49
14:13	1100W	$2x10^{-5}$	12.00	x	x	x	NC	45 ma	2.00A	2.23	12.79
14:30	1100W	$2x10^{-5}$	12.00	x	x	x	NC	45 ma	2.00A	2.32	12.81
14:33	1100W	$2x10^{-5}$	12.00	NC	x	x	29 ma	45 ma	NC	2.02	12.49
14:45	1100W	$2x10^{-5}$	12.00	NC	x	x	29 ma	46 ma	NC	1.81	12.49
14:50	1100W	$2x10^{-5}$	12.00	x	x	x	NC	45 ma	2.00A	NC	NC
14:55	1100W	$2x10^{-5}$	12.00	NC	x	x	NC	NC	NC	NC	NC
15:05											
15:08	1100W		12.00	x	x	x	NC	45 ma	2.00A	2.12	12.79
15:15											

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ble A-1
t No. 1 Telemetry Data

OLTAGE		TELEMETRY COMMUTATOR OUTPUT								CENTIGRADE TEMPERATURES REFER TO THERMISTORS
		LOW CALIB.	HIGH CALIB.	BATT. E_b	E_k	E_p	SKIN INSIDE	BATT.	SOLAR CELL	NOTES
BATT NO 1	BATT NO 2	NO 1	NO.2	NO 3	NO.4	NO 5	NO 6	NO 7	NO 8	
12.49	12.23	2.04	4.31	2.43	1.30	1.30	2.79	2.87	2.80	#6-30°, #8-33.5°
12.49	12.24	2.04	4.32	2.42	1.30	1.30	2.71	2.89	2.80	#6-33°, #8-34°
12.49	12.24	2.04	4.32	2.42	1.30	1.30	2.69	2.87	2.80	#6-33°, #8-33°
12.49	12.24	2.02	4.32	2.42	1.30	1.30	2.89	2.89	3.11	#6-30°, #8-27°
12.49	12.22	2.02	4.31	2.42	1.30	1.30	3.09	2.89	3.39	#6-23°, #8-20°
12.50	12.29	No Stepping --- All Readings 3.18 Volts								
12.52	12.31			No Change						
12.52	12.31			No Change						#6=4.5°, #8=16°
12.53	12.31			No Change						#6=9.5°, #8=23°
12.51	12.30			No Change						#6=13°, #8=25°
12.51	12.31			No Change		Now 3.20 Volts				#6=18°, #8=33°
12.51	12.31			No Change						#6=23°, #8=40°
12.50	12.30	2.02	4.29	2.47	1.31	1.31	2.39	3.55	2.43	
12.80	12.59	2.29	4.59	2.59	3.54	3.21	2.69	3.79	2.83	#6=32.5°, #8=37°
12.80	12.59	2.31	4.61	2.66	3.99	3.34	3.17	3.68	3.42	#6=27°, #8=24.5°
12.49	12.29	2.02	4.31	2.53	1.31	1.31	2.99	3.39	3.24	
12.49	12.29	2.01	4.30	2.54	1.31	1.31	3.33	3.33	3.54	#6=22°, #8=17°
12.79	12.59	2.28	4.57	2.66	4.02	3.39	3.71	3.60	3.92	Xpndr Oscillates abt. 3 min. at turn-on
12.78	12.57	2.27	4.56	2.70	4.02	3.45	3.94	3.58	4.12	
12.44	12.29	1.99	4.28	2.57	1.31	1.31	3.72	3.29	3.90	#6=17°, #8=10°
12.49	12.28	1.99	4.27	2.57	1.31	1.31	3.85	3.31	4.04	#6=15°, #8=7°
12.78	12.57	No Stepping --- All Readings 3.49 Volts								#6=12.5°, #8=6°
NC	NC	No Stepping --- All Readings 3.19 Volts								Xpndr "OFF"
12.79	12.57	No Stepping --- All Readings 3.49 Volts								#6=12.5°, #8=5°
12.49	12.28	No Stepping --- All Readings 3.19 Volts								NC
12.49	12.28	No Stepping --- All Readings 3.19 Volts								#6=13.5°, #8=13°
12.79	12.59	No Stepping --- All Readings 3.49 Volts								#6=13.5°, #8=13°
12.81	12.60	No Stepping --- All Readings 3.51 Volts								#6=17°, #8=21.5°
12.49	12.29	No Stepping -- All Readings 3.19 Volts								#6=17°, #8=21.5°
12.49	12.29	2.05	4.14	2.69	1.40	1.40	3.10	3.39	3.19	#6=20°, #8=28.5°
NC	NC	No Stepping --- All Readings 3.51 Volts								Xpndr "ON"
NC	NC	No Stepping --- All Readings 3.19 Volts								Xpndr "OFF"
		Stepping Returned Spontaneously								#6=25°, #8=36°
12.79	12.59	No Stepping --- All Readings 3.51 Volts								
		Turned off Xpndr, Stepping Returned								

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Table A-1
Geodetic Spacecraft No. 1

TIME	CHAMBER THERMAL CONDITION	CHAMBER PRESSURE mm Hg	BATT SUPPLY VOLTS	SELECT CALL OK	TM "ON" OK	TM "OFF" OK	CURRENT			VOLTAG	
							SECOR RCVR	TM	SECOR XPNDER+TM	SOLAR CELL	BATT NO 1
15:16	1100W	5.5×10^{-5}	12.00	x	x	x	NC	45 ma	2.00A	2.01	12.79
15:20	1100W	5.5×10^{-5}	12.00	NC	x	x	29 ma	46 ma	NC	NC	NC
15:35	Go cold	2×10^{-5}	12.00	NC	x	x	29 ma	47 ma	NC	0	12.49
15:40	Cold	2×10^{-5}	12.00	x	x	x	NC	46 ma	NC	0.26	12.72
15:50	Cold	2×10^{-5}	12.00	x	x	x	NC	45 ma	2.00A	0.21	12.79
15:55	Cold	2×10^{-5}	12.00		x	x	29 ma	47 ma	NC	0.04	12.48
16:10	Cold	1.8×10^{-5}	12.00	NC	x	x	29 ma	46 ma	NC	0.02	12.45
16:20	Cold	1.8×10^{-5}	12.00	x	x	x	NC	46 ma	2.00A	0.22	12.76
16:40	Cold	1.8×10^{-5}	12.00	x	x	x	NC	45 ma	2.00A	0.21	12.71
16:50	Cold	1.8×10^{-5}	12.00	NC	x	x	29 ma	45 ma	NC	0.02	12.42
17:05	Heat	1.7×10^{-5}	12.00	x	x	x	30 ma	46 ma	2.10A	2.32	12.74
17:15	Heat	1.7×10^{-5}	12.01	NC	x	x	30 ma	45 ma	NC	2.75	12.44
17:30	Heat	2.2×10^{-5}	12.00	x	x	x	28 ma	46 ma	2.00A	2.79	12.76
17:45	Heat	2.3×10^{-5}	12.09	NC	x	x	29 ma	46 ma		2.36	12.45
18:00	Heat	2.3×10^{-5}	12.00	x	x	x	29 ma	46 ma	2.10A	2.55	12.79
18:15	Heat	2.3×10^{-5}	12.00	NC	x	x	29 ma	46 ma		.89	12.45
18:22	Heat	2.6×10^{-5}	12.00	x	x	x	29 ma	46 ma		.05	12.45
18:30	Heat	2.6×10^{-5}	12.00	x	x	x	29 ma	46 ma	2.10A	.30	12.78
18:45	Heat	2.6×10^{-5}	11.81	x	x	x	29 ma	46 ma		.50	12.45
18:50	Heat	2.6×10^{-5}	12.01	x	x	x	29 ma	46 ma		1.73	12.44
18:56	Heat	2.6×10^{-5}	12.01	x	x	x	29 ma	46 ma	2.10A	.31	12.76
19:08	Cold	1.7×10^{-5}	12.01	x	x	x	29 ma	47 ma		.01	12.44
19:15	Cold	1.7×10^{-5}	12.00	x	x	x	29 ma	47 ma		.01	12.43
19:30	Cold	1.7×10^{-5}	12.00	x	x	x	29 ma	47 ma		.01	12.43
19:35	Cold	1.7×10^{-5}	12.00	x	x	x	29 ma	47 ma	2.10A	.01	12.75
19:45	Cold	1.7×10^{-5}	12.00	x	x	x	29 ma	46 ma		.01	12.42
20:00	Cold	1.7×10^{-5}	12.00	x	x	x	29 ma	46 ma	2.10A	.01	12.75
20:15	Cold	1.5×10^{-5}	12.00	x	x	x	29 ma	46 ma		.01	12.42
20:30	Cold	1.5×10^{-5}	12.00	x	x	x	29 ma	46 ma	2.10A	.01	12.75
20:40	Cold	1.6×10^{-5}	12.00	x	x	x	29 ma	46 ma		.01	12.44
20:45	Heat	1.6×10^{-5}	12.00	x	x	x	29 ma	47 ma	2.15A	4.03	12.74
20:55	Heat	1.6×10^{-5}	12.00	x	x	x	29 ma	46 ma		4.02	12.45
21:00	Heat	1.6×10^{-5}	12.00	x	x	x	29 ma	47 ma	2.10A	3.97	12.74
21:15	Heat	2.1×10^{-5}	12.00	x	x	x	29 ma	47 ma		3.29	12.45
21:20	Heat	2.1×10^{-5}	12.01	x	x	x	28 ma	46 ma	2.15A	3.50	12.75

A

ble A-1

t No. 1 Telemetry Data

OLTAGE		TELEMETRY COMMUTATOR OUTPUT								NOTES
		LOW CALIB.	HIGH CALIB.	BATT. E_b	E_k	E_p	SKIN INSIDE	BATT.	SOLAR CELL	CENTIGRADE TEMPERATURES REFER TO THERMISTORS
BATT NO. 1	BATT NO 2	NO. 1	NO. 2	NO. 3	NO. 4	NO. 5	NO 6	NO. 7	NO. 8	
12.79	12.59	2.51	4.32	2.76	3.99	3.35	3.04	3.59	3.01	#6=30°, #8=41°
NC	NC	2.11	4.20	2.60	1.41	1.41	2.60	3.24	2.59	#6=30°, #8=41°
12.49	12.29	2.10	4.25	2.59	1.39	1.39	2.61	3.19	2.79	#6=33°, #8=34.5°
12.72	12.52	2.39	4.49	2.85	4.10	3.38	3.07	3.42	3.29	NC
12.79	12.59	No Stepping --- All Readings 3.49 Volts								#6=30°, #8=26°
12.48	12.27	No Stepping --- All Readings 3.19 Volts								#6=24°, #8=20°
12.45	12.25	No Stepping --- All Readings 3.21 Volts								#6=22°, #8=13°
2.76	12.55	No Stepping --- All Readings 3.50 Volts								#6=22°, #8=13°
2.71	12.43	No Stepping --- All Readings 3.49 Volts								#6=17.5°, #8=8°
2.42	12.13	No Stepping --- All Readings 3.19 Volts								#6=17.5°, #8=8°
2.74	12.49	No Stepping --- All Readings 3.49 Volts								#6=14°, #8=5.5°
2.44	12.19	No Stepping --- All Readings 3.19 Volts								#6=16°, #8=12.5°
2.76	12.49	No Stepping --- All Readings 3.50 Volts								#6=20°, #8=17°
2.45	12.19	No Stepping --- All Readings 3.19Volts								#6=21°, #8=27°
2.79	12.50	No Stepping --- All Readings 3.52 Volts								#6=25°, #8=35°
2.45	12.19	No Stepping --- All Readings 3.21 Volts								#6=31°, #8=43°
2.45	12.19	2.21	4.15	2.50	1.52	1.52	2.44	3.10	2.52	#6=36°, #8=44°
2.78	12.49	2.84	4.15	3.09	3.85	3.41	3.05	3.44	3.08	#6=38°, #8=45°
2.45	12.19	2.56	3.81	2.81	1.99	1.99	2.65	3.09	2.81	#6=39°, #8=44°
2.44	12.19	2.61	3.76	2.89	2.08	2.08	2.69	3.09	2.79	#6=40°, #8=46°
2.76	12.49	2.64	4.40	2.84		3.30	2.74	3.52		#6=42°, #8=48°
2.44	12.19	2.60	3.75	2.89	2.32	3.22	Quit	Quit	Quit	#6=40°, #8=39°
2.43	12.19	No Stepping --- All Readings 3.23 Volts								#6=38°, #8=33.5°
2.43	12.19	No Stepping --- All Readings 3.22 Volts								#6=32°, #8=24°
2.75	12.49	No Stepping --- All Readings 3.53 Volts								#6=32°, #8=21°
2.42	12.19	No Stepping --- All Readings 3.21 Volts								#6=29°, #8=19.5°
2.75	12.49	No Stepping --- All Readings 3.53 Volts								#6=24°, #8=15.5°
2.42	12.19	No Stepping --- All Readings 3.52 Volts								#6=21.5°, #8=11°
2.75	12.49	No Stepping --- All Readings 3.50 Volts								#6=20°, #8=10°
2.44	12.19	No Stepping --- All Readings 3.20 Volts								#6=20°, #8= 8°
2.74	12.50	No Stepping --- All Readings 3.23 Volts								#6=18°, #8=13°
2.45	12.20	2.05	4.22	2.59	1.39	1.39	3.41	2.95	3.43	#6=20°, #8=20°
2.74	12.50	2.34	4.43	2.73	3.99	3.42	3.60	3.29	3.58	#6=22.5°, #8=22.5°
2.45	12.19	2.09	4.22	2.59	1.39	1.39	3.09	3.00	3.06	#6=25°, #8=28.5°
2.75	12.50	2.37	4.52	2.73	4 - 4.90	3.37	3.31	3.29	3.29	#6=28°, #8=32°

B

Table A-1
Geodetic Spacecraft No. 1

TIME	CHAMBER THERMAL CONDITION	CHAMBER PRESSURE mmHg	BATT SUPPLY VOLTS	SELECT CALL OK	TM "ON" OK	TM "OFF" OK	CURRENT			VOLTAGE	
							SECOR RCVR	TM	SECOR XPNDER+TM	SOLAR CELL	BATT NO 1
21:30	Heat	1.8×10^{-5}	12.02	x	x	x	29 ma	46 ma		2.86	12.45
21:45	Heat	2.2×10^{-5}	12.00	x	x	x	30 ma	46 ma	2.10A	2.99	12.74
22:00	Heat	3.4×10^{-5}	12.01	x	x	x	29 ma	46 ma		.73	12.43
22:15	Heat	1.8×10^{-5}	12.00	x	x	x	29 ma	46 ma	2.10A	2.66	12.76
22:20	Cold		12.00	x	x	x	30 ma	47 ma		.01	12.45
22:30	Cold	1.5×10^{-5}	12.01	x	x	x	30 ma	47 ma	2.10A	.01	12.73
22:37	Cold	1.5×10^{-5}	12.01	x	x	x	30 ma	47 ma		.01	12.43
22:45	Cold	1.6×10^{-5}	12.00	x	x	x	30 ma	47 ma	2.15A	.01	12.74
23:00	Cold	1.5×10^{-5}	12.00	x	x	x	30 ma	48 ma		.01	12.43
23:15	Cold	1.6×10^{-5}	12.00	x	x	x	30 ma	48 ma	2.15A	.01	12.74
23:30	Cold	1.7×10^{-5}	12.00	x	x	x	30 ma	48 ma		.01	12.43
23:45	Cold	1.6×10^{-5}	12.01	x	x	x	30 ma	48 ma	2.15A	.01	12.75
24:00	Cold	1.6×10^{-5}	12.00	x	x	x	30 ma	48 ma		.01	12.44
08:50	23°C	2×10^{-5}	12.03	x	x	x	NC	45 ma	2.00A	.20	12.69
08:55	23°C	2×10^{-5}	12.01	NC	x	x	29 ma	46 ma	NC	0	12.41
09:20	Hi heat	2×10^{-5}	12.00	NC	x	x	29 ma	46 ma	NC	9.62	12.43
09:25	3800W	2×10^{-5}	12.10	x	x	x	NC	45 ma	2.00A	9.40	12.79
09:50	Heat	2×10^{-5}	12.07	x	x	x	NC	46 ma	2.00A	0.85	12.79
09:55	Lo heat	2×10^{-5}	12.03	NC	x	x	29 ma	47 ma	NC	0.34	12.45
10:02	Lo heat	2.5×10^{-5}	12.02	NC	x	x	30 ma	47 ma	NC	0.30	12.49
10:07	Lo heat	2.5×10^{-5}	12.01	x	x	x	NC	47 ma	2.00A	0.29	12.79
10:25		2.5×10^{-5}	12.03	x	x	x	NC	47 ma	2.00A	0.30	12.76
10:30		2.5×10^{-5}	12.03	NC	x	x	30 ma	48 ma	NC	0.22	12.45
10:50		2.4×10^{-5}	12.00	NC	x	x	30 ma	48 ma	NC	0.09	12.45
10:52	Heat	2.4×10^{-5}	12.01	x	x	x	NC	47 ma	2.00A	0.31	12.75
11:05	Heat	2.4×10^{-5}	12.03	x	x	x	NC	47 ma	2.00A	0.35	12.75
11:10	Heat	2.4×10^{-5}	12.05	NC	x	x	29 ma	47 ma	NC	0.03	12.44
11:25	Heat	2.5×10^{-5}	12.00	NC	x	x	30 ma	48 ma	NC	0.05	12.43
11:30	Heat	2.5×10^{-5}	12.02	x	x	x	NC	47 ma	2.00A	0.30	12.75
11:45	Cold	2×10^{-5}	12.01	x	x	x	NC	47 ma	2.00A	0.01	12.43
11:50	Cold	2×10^{-5}	12.00	NC	x	x	30 ma	47 ma	NC	NC	NC
12:05	Cold	1.6×10^{-5}	12.00	NC	x	x	30 ma	47 ma	NC	NC	NC
12:10	Cold	1.6×10^{-5}	12.02	x	x	x	NC	47 ma	2.00A	0.24	12.72
12:25	Cold	1.7×10^{-5}	12.00	x	x	x	NC	46 ma	2.00A	0.22	12.72
12:30	Cold	1.7×10^{-5}	12.00	NC	x	x	30 ma	46 ma	NC	0	12.43

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A-1
o. 1 Telemetry Data

		LOW CALIB.	HIGH CALIB.	BATT. E_b	E_k	E_p	SKIN INSIDE	BATT.	SOLAR CELL	CENTIGRADE TEMPERATURES REFER TO THERMISTORS
TAGE		TELEMETRY COMMUTATOR OUTPUT								NOTES
TT O 1	BATT NO 2	NO 1	NO. 2	NO 3	NO. 4	NO 5	NO 6	NO. 7	NO. 8	
45	12.19	2.10	4.23	2.51	1.40	1.41	2.84	2.95	2.79	#6=29.5°,#8=35°
74	12.49	2.39	4.55	2.72	4 - 4.90	3.32	2.84	3.20	2.81	#6=32°,#8=42°
43	12.19	2.10	4.29	2.46	1.39	1.39	2.40	2.82	2.49	#6=37°,#8=43°
76	12.50	2.40	4.59	2.69	4 - 4.90	3.29	2.75	3.11	2.79	#6=40°,#8=40°
45	12.19	2.12	4.29	2.54	1.39	1.39	2.40	2.79	2.42	#6=42°,#8=45°
73	12.49	2.47	4.49	2.72	4 - 5.00	3.29	2.94	3.09	3.13	#6=37°,#8=35°
43	12.19	2.29	4.05	2.69	1.69	1.70	2.82	2.82	2.99	#6=34°,#8=27°
74	12.49	2.56	4.40	2.83	4 - 5.00	3.31	3.30	3.08	3.45	#6=32°,#8=23°
43	12.19	2.39	3.99	2.80	1.75	1.75	3.19	2.85	3.34	#6=29°,#8=19°
74	12.49	2.56	4.20	Quit	Quit	Quit	Quit	Quit	Quit	#6=25°,#8=14°
43	12.19	No Stepping --- All Readings 3.21 Volts								#6=23°,#8=11°
75	12.50	No Stepping --- All Readings 3.47 Volts								#6=19.5°,#8=7.5°
44	12.19	No Stepping --- All Readings 3.22 Volts								#6=18°, #8=6°
69	12.42	No Stepping --- All Readings 3.42 Volts								#6=21°, #8=23°
41	12.19	No Stepping -- All Readings 3.19 Volts								
43	12.19	No Stepping --- All Readings 3.19 Volts								#6=21°, #8=24°
79	12.50	No Stepping --- All Readings 3.49 Volts					(Heat)			#6=21°, #8=28°
79	12.51	No Stepping --- All Readings 3.51 Volts (#6 TC intermittently open)								#6=32°,#8=39°
45	12.20	No Stepping --- All Readings 3.19 Volts								#6=25°,#8=39°
.49	12.20	2.29	4.09	2.69	1.60	1.62	2.59	3.10	2.69	#6=33°,#8=38.5°
.79	12.50	2.59	4.39	2.84	4.25	3.33	2.90	3.39	3.00	#6=34.5°,#8=39°
.76	12.52	2.62	4.37	2.88	4.35	3.35	2.92	3.35	3.01	#6=36°, #8=38°
.45	12.21	2.34	4.05	2.70	1.65	1.67	2.63	3.04	2.71	#6=36°, #8=38°
.45	12.20	2.36	4.03	2.71	1.70	1.73	2.6[illegible]	2.99	2.72	#6=35°, #8=38°
.75	12.51	2.67	4.33	2.89	4.23	3.32	2.93	3.29	3.01	#6=35°, #8=38°
.75	12.51	2.64	4.33	2.91	4.37	3.34	2.90	3.23	2.97	#6=37°, #8=40.5°
.44	12.20	2.29	4.13	2.67	1.59	1.60	2.53	2.89	2.61	#6=39°, #8=41.5°
.43	12.19	2.23	4.21	2.61	1.50	1.51	2.43	2.80	2.53	#6=40°, #8=41
.75	12.50	2.54	4.49	2.80	4.39	3.30	2.73	3.10	2.84	#6=40°, #8=41°
2.43	12.19	2.62	4.40	2.84	4.19	3.32	3.04	3.08	3.20	#6=37°, #8=31°
C	NC	2.56	3.70	2.94	2.19	2.19	3.02	2.97	3.13	#6=37°, #8=31°
C	NC	2.78	3.62	2.98	2.44	2.40	3.16	3.03	3.22	#6=32°, #8=22°
2.72	12.49	2.95	4.05	3.11	3.98	3.43	3.46	3.26	3.55	#6=32°, #8=22°
2.72	12.48	2.92	4.01	3.12	4.02	3.44	3.55	3.29	3.64	#6=26°, #8=15°
2.43	12.19	2.79	3.49	3.04	2.60	2.60	3.29	3.05	3.29	#6=26°, #8=15°

B

Table A-1
Geodetic Spacecraft No. 1

TIME	CHAMBER THERMAL CONDITION	CHAMBER PRESSURE mmHg	BATT SUPPLY VOLTS	SELECT CALL OK	TM "ON" OK	TM "OFF" OK	CURRENT			VOLTAGE	
							SECOR RCVR	TM	SECOR XPNDER+TM	SOLAR CELL	BATT NO 1
12:45	Cold	$1.7x10^{-5}$	12.01	NC	x	x	30 ma	46 ma	NC	0	12.42
12:50	Cold	$1.7x10^{-5}$	12.00	x	x	x	NC	46 ma	2.00A	0.21	12.72
13:00	Cold	$1.7x10^{-5}$	12.01	x	x	x	NC	46 ma	2.00A	0.22	12.73
13:05	Cold	$1.7x10^{-5}$	12.00	NC	x	x	30 ma	46 ma	NC	0	12.42
13:15	Heat	$1.7x10^{-5}$	12.00	NC	x	x	30 ma	46 ma	NC	5.20	12.42
13:20	Heat	$1.7x10^{-5}$	12.00	x	x	x	NC	47 ma	2.00A	3.70	12.73
13:30	Heat	$1.7x10^{-5}$	12.00	x	x	x	NC	47 ma	2.00A	3.61	12.73
13:35	Heat	$1.7x10^{-5}$	12.01	NC	x	x	30 ma	46 ma	NC	3.19	12.42
13:45	Heat	$1.7x10^{-5}$	12.00	NC	x	x	30 ma	46 ma	NC	2.82	12.42
13:50	Heat	$1.9x10^{-5}$	12.00	x	x	x	NC	46 ma	2.00A	2.99	12.73
14:00	Heat	$1.9x10^{-5}$	12:01	x	x	x	NC	46 ma	2.00A	2.79	12.73
14:05	Heat	$1.9x10^{-5}$	12.00	NC	x	x	30 ma	46 ma	NC	2.25	12.42
14:15	Heat.	$2x10^{-5}$	12.01	NC	x	x	30 ma	46 ma	NC	1.69	12.43
14:20	Heat	$2x10^{-5}$	12.00	x	x	x	NC	46 ma	2.00A	1.21	12.73
14:30	Heat	$2x10^{-5}$	12.00	x	x	x	NC	46 ma	2.00A	.29	12.73
14:35	Heat	$2x10^{-5}$	12.00		IMPOSSIBLE TO TURN OFF THE SECOR TRANSMITTER						
14:45	Heat	$4.1x10^{-5}$	12.01	NC	x	x	30 ma	46 ma	NC	.52	12.43
14:50	Heat	$4.1x10^{-5}$	12.00	x	x	x	NC	46 ma	2.00A	.81	12.74
14:55	Heat		12.00	x	x	x	NC	46 ma	2.00A	.50	12.74
15:00	Heat off		12.00	NC	x	x	30 ma	46 ma	NC	.01	12.43
15:10	Room	$3x10^{-5}$	12.00	NC	x	x	30 ma	46 ma	NC	.01	12.43
15:15	Room	$3x10^{-5}$	12.00	x	x	x	NC	46 ma		.04	12.73
15:20	Room	$3x10^{-5}$	12.00	NC	x	x	30 ma	46 ma	NC	.04	12.42
15:25	Room		12.00	NC	x	x	30 ma	46 ma	NC	.01	12.42
15:30	Room		12.00	NC	x	x	30 ma	46 ma	NC	.01	12.42
15:45	Room	Sea Level	12.00	NC	x	x	30 ma	46 ma	NC	.01	12.43
15:50	Room	Sea Level	12.00	x	x	x	NC	46 ma	2.00A	.01	12.73

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le A-1
No. 1 Telemetry Data

		LOW CALIB.	HIGH CALIB.	BATT. E_b	E_k	E_p	SKIN INSIDE	BATT.	SOLAR CELL	CENTIGRADE TEMPERATURES REFER TO THERMISTORS
LTAGE		TELEMETRY COMMUTATOR OUTPUT								NOTES
BATT NO 1	BATT NO 2	NO 1	NO 2	NO 3	NO 4	NO 5	NO 6	NO 7	NO 8	
.42	12.19	2.79	3.53	3.01	2.53	2.50	3.29	3.05	3.33	#6=23°, #8=12°
.72	12.49	2.94	4.02	3.09	3.99	3.43	3.62	3.30	3.70	#6=23°, #8=12°
.73	12.49	2.71	4.19	3.01	3.99	3.42	3.70	3.20	3.82	#6=22°, #8=11°
.42	12.19	2.42	3.89	2.82	2.05	2.05	3.49	2.93	3.59	#6=22°, #8=11°
.42	12.19	2.42	3.89	2.79	1.92	1.92	3.50	2.95	3.59	#6=19.5°, #8=8°
.73	12.49	2.73	4.19	2.99	4.11	3.40	3.69	3.29	3.70	#6=19.5°, #8=8°
.73	12.49	2.65	4.29	2.99	4.30	3.49	3.69	3.24	3.65	#6=21°, #8=18°
.42	12.19	2.39	3.92	2.69	1.79	1.79	3.29	2.94	3.29	#6=21°, #8=18°
.42	12.19	2.36	3.95	2.75	1.75	1.75	3.19	2.99	3.19	#6=25°, #8=25°
.73	12.49	2.69	4.22	2.91	4.16	3.39	3.43	3.24	3.43	#6=25°, #8=25°
.73	12.50	2.62	4.31	2.89	4.20	3.39	3.30	3.23	3.29	#6=27.5°, #8=31°
.42	12.19	2.39	4.03	2.69	1.69	1.69	2.96	2.96	2.97	#6=27.5°, #8=31°
.43	12.19	2.32	4.04	2.69	1.69	1.69	2.85	2.92	2.90	#6=32°, #8=35°
.73	12.49	2.62	4.33	2.89	4.20	3.34	3.12	3.22	3.15	#6=32°, #8=35°
.73	12.49	2.63	4.32	2.87	4.31	3.32	3.20	3.05	3.01	#6=33.5°, #8=36.5°
										#6=33.5°, #8=36.5°
.43	12.19	2.39	4.02	2.71	1.70	1.71	2.69	2.89	2.72	#6=36°, #8=41°
.74	12.49	2.61	4.35	2.86	4.25	3.32	2.94	3.15	3.00	#6=36°, #8=41°
.74	12.49	2.63	4.39	2.82	4.15	3.31	2.90	3.13	2.94	#6=37°, #8=42°
.43	12.19	2.39	4.09	2.69	1.69	1.69	2.61	2.85	2.69	
.43	12.19	2.39	4.09	2.69	1.69	1.69	2.64	2.82	2.75	#6=37°, #8=37°
.73	12.49	Altitude Brought Down to Sea Level								
.42	12.19	2.39	4.03	2.70	1.72	1.72	2.72	2.83	2.85	#6=36°, #8=33.5°
.42	12.19	2.39	4.09	2.70	1.72	1.72	2.79	2.82	2.88	#6=35°, #8=32°
.42	12.19	2.39	4.09	2.70	1.71	1.71	2.79	2.83	2.92	#6=35°, #8=32°
.43	12.20	2.37	4.01	2.63	1.69	1.69	2.79	2.79	2.90	#6=34°, #8=32°
.73	12.49	No Stepping --- All Readings 3.52 Volts								

B

APPENDIX B

GEODETIC SPACECRAFT NO. 2

TELEMETRY DATA

Table B-1
Geodetic Spacecraft No. 2 [illegible]

TIME	CHAMBER PRESSURE mmHg	SECOR ON = x	TM "ON" OK	TM "OFF" OK	CURRENT SECOR XPNDER+TM	CURRENT TM			TELEMETR NO 1 LOW CALIB.	NO 2 HIGH CALIB.	NO 3 BATT. E_b
17:00	Start	x	x	x					2.01	3.99	3.11
17:30	Test	x	x	x	1.85A				2.25	4.21	3.29
11:00	$1.5x10^{-5}$		x	x		79 ma			1.99	3.93	3.19
11:30	$5x10^{-6}$		x	x		78 ma			2.01	3.99	3.11
12:00	$4.5x10^{-6}$		x	x		78 ma			1.99	3.99	3.12
12:15	$4.5x10^{-6}$	x	x	x	1.80A	75 ma	Note: 12.19V DC		2.25	4.21	3.29
12:30	$4.4x10^{-6}$		x	x		78 ma	Note: 12.33V DC		1.99	3.92	3.22
12:45	$4.3x10^{-6}$	x	x	x	1.85A	78 ma			2.24	4.19	3.39
13:00	$4x10^{-6}$		x	x		78 ma			1.99	3.89	3.30
13:15		x	x	x	1.90A	78 ma	Scope shows spikes on #4		2.24	4.14	3.35
13:30	$4x10^{-6}$		x	x		78 ma	Stepping skipped channels		1.96	3.85	3.29
NOTE:			Difficulty in manual stepping (skips channels); Running magnetic shaker next								
			by using very short tap on button -- extra spikes from shaker getting through								
13:45	$4x10^{-6}$	x	x	x	1.90A	78 ma	TM ok - shaker off		2.23	4.10	3.32
14:00	$4x10^{-6}$		x	x		75 ma			1.93	3.81	3.29
		Stop	Cold		Start	Hot					
14:15	$4.8x10^{-6}$	x	x	x	1.90A	75 ma			2.21	4.09	3.29
14:30	$5.5x10^{-6}$	x	x	x	1.90A		Shaker On		2.21	4.09	3.29
14:45	$1.6x10^{-5}$		x	x		78 ma			1.94	3.89	3.29
15:00	$1.2x10^{-5}$		x	x		78 ma			2.00	3.89	3.29
15:15	$1.3x10^{-5}$		x	x		76 ma			2.00	3.89	3.29
15:20	$1.3x10^{-5}$	x	x	x	1.90A				2.29	4.29	3.29
15:30	$1.2x10^{-5}$		x	x		78 ma			2.01	4.02	3.29
15:45	$1x10^{-5}$	x	x	x	1.90A				2.29	4.31	3.29
16:10			x	x		78 ma			2.02	4.01	3.29
08:40	$8x10^{-6}$		x	x		79 ma			1.99	3.89	3.29
09:00	$8x10^{-6}$		x	x		78 ma	Cold run starts		1.99	3.89	3.29
09:15	$3.8x10^{-6}$		x	x		78 ma	The shaker is on and it is impossible to rea				
09:20	$3.8x10^{-6}$		x	x		78 ma			1.99	3.92	3.29
09:30	$4.1x10^{-6}$		x	x		78 ma			1.99	3.90	3.29
09:35	$4x10^{-6}$	x	x	x	1.90A				2.24	4.19	3.39
09:45	$3.9x10^{-6}$		x	x		78 ma			1.99	3.91	3.29
10:00	$3.7x10^{-6}$	x	x	x	1.90A				2.24	4.19	3.39
10:15	$3.6x10^{-6}$		x	x		78 ma			1.99	3.89	3.29

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B-1
o. 2 Telemetry Data

TT. E_k E_p SKIN INSIDE BATT. SOLAR CELL

METRY COMMUTATOR OUTPUT						THERMOCOUPLES, TEMP IN °C				NOTES
.3	NO. 4	NO 5	NO 6	NO 7	NO 8	1	2	6	16	
1	1.49	1.59	3.22	3.24	3.17	27°	27°	27°	27°	No crystal in SC
9	2.51	3.30	3.50	3.48	3.50	27°	27°	27°	27°	Manual stepping erratic due to pickup in line
9	1.49	1.59	3.11	3.13	3.10	27°	27°	27°	27°	
1	1.49	1.56	3.39	3.19	3.36	25°	23°	19°	17°	
2	1.49	1.56	3.73	3.20	3.73	20°	16°	10°	8°	
9	2.51	3.30	4.19	3.49	4.19	20°	13°	6°	3°	
2	1.49	1.56	4.03	3.29	3.99	18°	11°	4°	2°	
9	2.62	3.39	4.39	3.59	4.39	15°	7°	0°	-2°	
0	1.49	1.56	4.24	3.39	4.22	14°	5°	-2°	-4°	
5	2.70	3.39	4.55	3.67	4.49	12°	4°	-4°	-8°	
9	1.49	1.56	4.39	3.49	4.31	11°	3°	-7°	-10°	
next to test;	Could s	tep on al	l channel	s						
rou	gh when b	utton he	ld down.							
2	2.68	3.39	4.63	3.72	4.59	8°	-3°	-10°	-14°	
9	1.49	1.55	4.49	3.52	4.39	6°	-5°	-13°	-16°	
9	2.59	3.30	4.64	3.79	4.50	10°	0°	-9°	-6°	
9	2.49	3.29	4.49	3.79	4.20	10°	5°	0°	2°	
9	1.49	1.59	3.61	3.69	3.32	13°	10°	12°	16°	
9	1.49	1.59	3.09	3.55	2.71	20°	23°	34°	38°	
9	1.49	1.59	2.61	3.59	2.49	23°	30°	42°	44°	
9	2.69	3.22	2.89	3.79	2.79	23°	30°	42°	44°	
9	1.49	1.59	2.39	3.49	2.29	30°	40°	50°	52°	
9	2.70	3.22	2.32	3.69	2.32	33°	43°	51°	54°	
9	1.49	1.59	2.38	3.50	2.30	?	?	?	?	
9	1.49	1.59	3.31	3.35	3.30	22°	22°	22°	22°	
9	1.49	1.59	3.32	3.34	3.30	22°	22°	22°	22°	
to r	ead TM					22°	21°	20°	17°	
9	1.49	1.59	3.49	3.40	3.47	20°	20°	16°	15°	
9	1.49	1.59	3.59	3.39	3.59	20°	17°	14°	11°	
9	2.85	3.42	3.91	3.60	3.91	19°	17°	11°	11°	
9	1.49	1.59	3.89	3.39	3.83	18°	15°	9°	5°	
9	2.84	3.43	4.29	3.62	4.29	16°	12°	6°	3°	
9	1.49	1.59	4.10	3.39	4.09	14°	9°	0°	-3°	

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Table B-1
Geodetic Spacecraft No. 2

TIME	CHAMBER PRESSURE mmHg	SECOR ON = X	TM "ON" OK	TM "OFF" OK	CURRENT				TELEMET		
					SECOR XPNDER+TM	TM			NO 1 LOW CALIB.	NO 2 HIGH CALIB.	NO 3 BATT. E_b
10:30	$3.6x10^{-6}$	x	x	x	1.95A				2.24	4.18	3.39
10:45	$4x10^{-6}$		x	x		77 ma			1.99	3.88	3.29
11:00	$3.6x10^{-6}$	x	x	x	2.00A				2.24	4.12	3.39
11:15	$3.5x10^{-6}$		x	x		77 ma			1.99	3.87	3.25
11:30	$3.6x10^{-6}$	x	x	x	1.95A				2.24	4.10	3.39
11:45	$3.6x10^{-6}$		x	x		76 ma			1.96	3.79	3.22
12:00	$3.6x10^{-6}$	x	x	x	1.95A				2.29	4.09	3.38
12:05	$3.6x10^{-6}$		x	x		77 ma			1.95	3.79	3.23
12:15	$3.8x10^{-6}$		x	x		76 ma			1.95	3.79	3.19
12:30	$4.4x10^{-6}$	x	x	x	1.95A				2.22	4.09	3.33
12:45	$5.1x10^{-6}$		x	x		76 ma			1.95	3.79	3.29
13:00	$6.6x10^{-6}$	x	x	x	1.95A				2.25	4.13	3.34
13:15	$7.7x10^{-6}$		x	x		77 ma			2.01	3.89	3.20
13:30	$1.2x10^{-5}$	x	x	x	1.95A				2.29	4.23	3.33
13:45	$3.8x10^{-6}$	x	x	x		76 ma			2.02	3.99	3.19
14:00	$3.4x10^{-6}$	x	x	x	1.95A				2.29	4.29	3.32
14:15	$3.6x10^{-6}$	x	x	x		76 ma			2.02	3.99	3.29
14:30	$3.4x10^{-6}$	x	x	x	1.95A				2.29	4.24	3.33
14:45	$3.4x10^{-6}$	x	x	x		76 ma			1.99	3.95	3.29
15:00	$3.4x10^{-6}$	x	x	x	1.90A				2.26	4.20	3.31
15:15			TM works on automatic but not manual because of interference; Turned heat lamps of								
15:42									1.99	3.89	3.29
15:45					1.90A				2.29	4.19	3.29
16:00									1.99	3.89	3.19
16:15									1.99	3.96	3.26
08:45	$2.7x10^{-5}$		x	x		78 ma			2.02	3.99	3.26
09:00	$3.5x10^{-6}$	x	x	x	1.95A				2.28	4.29	3.39
09:15	$3.4x10^{-6}$		x	x		78 ma			2.03	3.99	3.29
09:30	$3.4x10^{-6}$	x	x	x	1.95A				2.29	4.29	3.39
09:45	$3.4x10^{-6}$		x	x		77 ma			1.99	3.99	3.29
10:00	$3.3x10^{-6}$	x	x	x	1.95A				2.29	4.22	3.39
10:15	$3.4x10^{-6}$		x	x		77 ma			1.99	3.97	3.28
10:30	$3.4x10^{-6}$	x	x	x	1.95A				2.29	4.21	3.39
10:45	$3.2x10^{-6}$		x	x							

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able B-1
ft No. 2 Telemetry Data

BATT. E_b	E_k	E_p	SKIN INSIDE	BATT.	SOLAR CELL					
LEMETRY COMMUTATOR OUTPUT						THERMOCOUPLES, TEMP IN °C				NOTES
NO.3	NO.4	NO.5	NO.6	NO.7	NO.8	1	2	6	16	
3.39	2.79	3.42	4.49	3.67	4.43	12°	6°	-3°	-6°	
3.29	1.49	1.56	4.29	3.49	4.24	10°	3°	-5°	-8°	
3.39	2.79	3.42	4.59	3.72	4.53	9°	2°	-7°	-9°	
3.25	1.49	1.56	4.42	3.52	4.35	7°	1°	-10°	-12°	
3.39	2.73	3.39	4.70	3.81	4.64	5°	-2°	-11°	-15°	
3.22	1.49	1.56	4.49	3.59	4.41	4°	-3°	-13°	-15°	
3.38	2.71	3.89	4.69	3.89	4.69	3°	-5°	-14°	-17°	Heat Starts
3.23	1.49	1.56	4.53	3.69	4.45	4°	-4°	-12°	-12°	
3.19	1.49	1.56	4.52	3.69	4.42	4°	-3°	-9°	-9°	
3.33	2.71	3.35	4.60	4.01	4.39	6°	2°	0°	2°	
3.29	1.49	1.56	3.93	3.74	3.62	10°	10°	15°	18°	
3.34	2.69	3.34	3.73	3.99	3.51	14°	18°	26°	28°	
3.20	1.49	1.56	3.02	3.75	2.83	18°	25°	34°	37°	
3.33	2.64	3.29	3.02	3.99	2.89	23°	30°	40°	44°	Cold Starts
3.19	1.49	1.56	2.71	3.69	2.69	28°	34°	37°	38°	
3.32	2.69	3.29	3.35	3.86	3.39	25°	28°	30°	28°	
3.29	1.49	1.56	3.29	3.57	3.33	25°	25°	24°	21°	
3.33	2.69	3.30	3.81	3.75	3.84	23°	20°	20°	16°	
3.29	1.49	1.56	3.73	3.49	3.73	19°	18°	14°	10°	
3.31	2.65	3.29	4.19	3.69	4.12	18°	14°	10°	7°	Heat Starts
lamps off -- TM normal (Power connection had been moved to same circuit as Heat)										
3.29	1.39	1.49	3.59	3.39	3.39					
3.29	2.59	3.19	3.79	3.59	3.59	22°	22°	23°	20°	
3.19	1.39	1.49	3.19	3.29	3.00	25°	25°	27°	28°	
3.26	1.46	1.56	2.97	3.26	2.75	28°	30°	34°	38°	
3.26	1.49	1.58	3.23	3.27	3.20	27°	27°	27°	27°	Cold Starts
3.39	2.69	3.39	3.56	3.54	3.56	26°	25°	26°	24°	
3.29	1.49	1.56	3.49	3.29	3.50	25°	22°	21°	16°	
3.39	2.69	3.39	3.99	3.49	3.99	22°	18°	13°	12°	
3.29	1.49	1.56	3.89	3.29	3.82	22°	18°	12°	7°	
3.39	2.69	3.39	4.29	3.49	4.21	21°	14°	7°	4°	
3.28	1.49	1.59	4.11	3.29	4.09	19°	10°	3°	2°	
3.39	2.69	3.39	4.49	3.59	4.40	16°	9°	0°	-4°	
						14°	6°	-2°	-5°	

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Table B-
Geodetic Spacecraft No. 2

TIME	CHAMBER PRESSURE mmHg	SECOR ON = x	TM "ON" OK	TM "OFF" OK	CURRENT SECOR XPNDER+TM	CURRENT TM			TELEMET NO 1 LOW CALIB.	NO 2 HIGH CALIB.	NO 3 BATT. E_b
11:00	$3.2x10^{-6}$				1.95A				2.25	4.15	3.36
11:15	$3.2x10^{-6}$	x	x	x		77 ma			1.96	3.86	3.29
11:30	$3.4x10^{-6}$	x	x	x	1.95A				2.23	4.12	3.33
11:45	$3.4x10^{-6}$	x	x	x		77 ma			1.95	3.83	3.30
12:00	$3.4x10^{-6}$								1.94	3.81	3.31
12:15	$3.3x10^{-6}$								1.93	3.79	3.30
12:30	$3.4x10^{-6}$								1.92	3.79	3.30
12:45	$3.4x10^{-6}$								1.91	3.74	3.29
13:00	$3.2x10^{-6}$								1.91	3.72	3.29
13:15	$3.3x10^{-6}$								1.91	3.70	3.29
13:30	$3.4x10^{-6}$								1.89	3.69	3.29
			Skipping around - 4, 6, 7, 8, 2, 4, 6, 7, 8, 1, 2, 3, 4, 5, 6, 7, 8, 1 (Unobserved probably str								
13:45	$3.4x10^{-6}$								1.89	3.70	3.29
14:00	$3.6x10^{-6}$	x	x	x	1.95A				2.20	4.02	3.30
14:15	$7x10^{-6}$	x	x	x	1.95A				2.22	4.09	3.29
14:30	$7.6x10^{-6}$	x	x	x	1.95A				2.24	4.19	3.23
14:45	$9.5x10^{-6}$	x	x	x	1.95A				2.29	4.29	3.22
15:00	$8.7x10^{-6}$	x	x	x	1.95A				2.34	4.39	3.23
15:10					1.95A				2.36	4.46	3.24
15:11						78 ma			2.08	4.19	3.19
15:15	$4.2x10^{-6}$	x	x	x		78 ma			2.09	4.32	3.21
15:30	$4x10^{-6}$	x	x	x					2.10	4.25	3.24
15:45	$3.4x10^{-6}$	x	x	x					2.09	4.22	3.26
16:00	$3.2x10^{-6}$	x	x	x					2.09	4.19	3.27
08:15	$1.1x10^{-5}$		x	x		78 ma			2.01	3.87	3.24
08:30	$4x10^{-6}$		x	x		78 ma			2.02	3.91	3.26
08:45	$3.3x10^{-6}$		x	x		78 ma			2.01	3.92	3.28
09:00	$3.2x10^{-6}$		x	x		78 ma			2.00	3.92	3.29
09:15	$3.2x10^{-6}$		x	x		78 ma			2.00	3.91	3.29
09:30	$3.4x10^{-6}$	x	x	x	1.95A				2.24	4.19	3.39
09:45	$3.2x10^{-6}$		x	x		78 ma			2.00	3.89	3.29
10:00	$3x10^{-6}$	x	x	x	1.95A				2.23	4.19	3.39
10:15	$3.1x10^{-6}$		x	x		78 ma			1.99	3.89	3.29
10:30	$3.4x10^{-6}$	x	x	x	1.95A				2.23	4.10	3.39

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e B-1
No. 2 Telemetry Data

ATT. E_b	E_k	E_p	SKIN INSIDE	BATT.	SOLAR CELL					
EMETRY COMMUTATOR OUTPUT						THERMOCOUPLES, TEMP IN °C				NOTES
NO.3	NO.4	NO 5	NO 6	NO 7	NO 8	1	2	6	16	
.36	2.64	3.39	4.62	3.65	4.55	12°	4°	-6°	-8°	
.29	1.49	1.58	4.42	3.44	4.34	9°	1°	-9°	-12°	
.33	2.63	3.34	4.69	3.70	4.61	9°	1°	-10°	-13°	
.30	1.49	1.56	4.51	3.52	4.43	7°	-1°	-11°	-14°	
.31	1.49	1.56	4.55	3.59	4.49	7°	-3°	-13°	-16°	
.30	1.49	1.56	4.59	3.63	4.50	5°	-4°	-13°	-17°	
.30	1.49	1.56	4.59	3.68	4.52					
.29	1.49	1.56	4.62	3.73	4.54	0°	-8°	-18°	-20°	
.29	1.49	1.55	4.62	3.79	4.55	-1°	-10°	-20°	-21°	Heat On
.29	1.49	1.56	4.60	3.84	4.51	-2°	-9°	-17°	-16°	
.29	1.49	1.55	4.52	3.89	4.39	-4°	-7°	-11°	-10°	
ly stray interference)										Heat Lamps
.29	1.49	1.55	4.33	3.95	4.12	0°	-2°	-1°	2°	
.30	2.49	3.29	4.34	4.21	4.08	3°	4°	10°	12°	
.29	2.42	3.19	3.43	4.19	3.09	10°	17°	27°	34°	Interference from Heat Lamps
.23	2.42	3.19	3.01	4.19	2.79	18°	25°	42°	48°	
.22	2.41	3.11	2.53	4.12	2.40	27°	45°	57°	60°	
.23	2.44	3.11	2.27	4.03	2.21	39°	53°	66°	69°	
.24	2.42	3.09	2.19	3.96	2.13					Cold On
.19	1.47	1.59	1.89	3.69	1.84					TM Off
.21	1.47	1.59	1.88	3.62	1.89	44°	58°	67°	66°	
.24	1.46	1.59	2.09	3.47	2.14	46°	50°	56°	55°	
.26	1.46	1.59	2.43	3.30	2.52	42°	42°	46°	44°	
.27	1.46	1.59	2.69	3.19	2.79	40°	40°	36°	35°	
.24	1.49	1.59	3.39	3.49	3.37	19°	19°	19°	19°	Cold Starts
.26	1.49	1.59	3.59	3.51	3.58	19°	19°	18°	14°	
.28	1.49	1.59	3.80	3.51	3.79	18°	14°	10°	10°	
.29	1.49	1.59	3.93	3.52	3.90	16°	10°	5°	3°	
.29	1.49	1.58	4.09	3.49	4.03	13°	9°	4°	1°	
.39	2.69	3.39	4.45	3.79	4.39	12°	5°	0°	-4°	
.29	1.49	1.56	4.29	3.52	4.20	10°	3°	-4°	-7°	
.39	2.69	3.39	4.60	3.80	4.53	9°	1°	-6°	-8°	Heat Starts
.29	1.49	1.56	4.41	3.59	4.29	11°	4°	-4°	-4°	
.39	2.65	3.35	4.50	3.85	4.29	11°	6°	3°	3°	

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Table B-
Geodetic Spacecraft No. 2

TIME	CHAMBER Pressure mm Hg	Secor ON: x	TM "ON" OK	TM "OFF" OK	CURRENT Secor xpndr+TM	CURRENT TM			TELEMETRY No. 1 LOW CALIB.	No. 2 HIGH CALIB.	No. 3 BATT. E_b
10:45	3.3×10^{-6}		x	x		78 ma			1.99	3.82	3.19
11:00	4.4×10^{-6}	x	x	x	1.95A				2.24	4.19	3.32
11:15	4.8×10^{-6}		x	x		77 ma			1.99	3.89	3.19
11:30	6×10^{-6}	x	x	x	1.90A				2.32	4.29	3.35
11:45	3.4×10^{-6}	x	x						2.03	4.05	3.29
12:00	3.2×10^{-6}	x	x	x	1.90A				2.32	4.36	3.30
12:15	3.1×10^{-6}	x	x	x					2.04	4.09	3.26
12:30	3×10^{-6}	x	x	x					2.31	4.34	3.29
12:45	3×10^{-6}										
13:00	3.1×10^{-6}								2.03	4.05	3.29
13:15	3.1×10^{-6}	x	x	x	1.90A				2.31	4.33	3.29
13:30									2.03	4.09	3.24

A

ble B-1
t No. 2 Telemetry Data

ATT. E_b	E_k	E_p	SKIN INSIDE	BATT.	SOLAR CELL					
METRY COMMUTATOR OUTPUT						THERMOCOUPLES, TEMP IN °C				NOTES
No. 3	No. 4	No. 5	No. 6	No. 7	No. 8	1	2	6	16	
3.19	1.49	1.56	3.99	3.60	3.69	13°	11°	11°	14°	
3.32	2.61	3.29	3.85	3.85	3.62	17°	17°	22°	22°	
3.19	1.49	1.59	3.24	3.59	3.01	20°	26°	33°	38°	
3.35	2.58	3.24	2.88	3.84	2.69	26°	36°	45°	50°	Cold Starts
3.29	1.49	1.59	2.40	3.57	2.34	30°	40°	48°	47°	
3.30	2.55	3.21	2.96	3.74	3.03	32°	36°	39°	38°	
3.26	1.46	1.59	2.95	3.41	3.01	31°	31°	32°	31°	
3.29	2.52	3.20	3.53	3.55	3.55	29°	26°	24°	21°	Heat Starts
						28°	25°	20°	17°	
3.29	1.46	1.59	3.33	3.21	3.10	29°	25°	24°	25°	
3.29	2.55	3.21	3.29	3.43	3.08	30°	30°	30°	32°	
3.24	1.46	1.59	2.75	3.19	2.54	32°	35°	42°	41°	

B

APPENDIX C

GEODETIC SPACECRAFT NO. 3

TELEMETRY DATA

Table C-1
Geodetic Spacecraft No. 3

TIME	CHAMBER THERMAL CONDITION	CHAMBER PRESSURE mmHg	BATT SUPPLY VOLTS	SELECT CALL OK	TM "ON" OK	TM "OFF" OK	CURRENT			VOLTAGE	
							Select Call on	Select Call off	SECOR XPNDER+TM	SOLAR CELL	BATT NO 1
					x			78 ma			
09:00					x	x	1.85A				
10:30					x			79 ma			
10:35					x	x	1.85A				
11:30					x			79 ma			
11:35					x	x	1.85A				
12:00					x			78 ma			
12:05					x	x	1.85A	78 ma			
13:05					x			78 ma			
13:30					x	x	1.85A	78 ma			
13:35					x			78 ma			
14:00					x	x	1.85A	78 ma			
14:15					x			78 ma			
14:30					x	x	1.85A	78 ma			
15:00								78 ma			
					x			78 ma			
16:30											
18:00											
19:38										19V	
19:45					x	x				19.5V	
00:25					x	x		78 ma			
00:45					x			78 ma			
01:15					x			78 ma			
01:45					x			78 ma			
02:00					x			78 ma			
02:15					x					17.5V	
02:45					x						
03:00											

A

Table C-1
Aft No. 3 Telemetry Data

VOLTAGE BATT NO 1	BATT NO 2	NO 1 LOW CALIB.	NO. 2 HIGH CALIB.	NO 3 BATT. E_b	NO. 4 E_k	NO 5 E_p	NO 6 SKIN INSIDE	NO 7 BATT.	NO 8 SOLAR CELL	NOTES (TEMPERATURES REFER TO THERMISTORS)
		1.99	3.99	3.09	1.19	1.29	2.99	3.09	3.19	
		2.29	4.21	3.09	2.69	2.89	3.29	3.39	3.30	
		1.99	3.99	3.04	1.19	1.29	3.29	3.03	3.62	
		2.29	4.28	3.06	2.69	2.96	3.59	3.39	3.89	
		1.99	3.90	3.16	1.22	1.39	3.79	3.22	3.90	
		2.19	4.09	2.99	2.89	3.39	3.99	3.39	4.39	
		1.99	3.79	3.03	1.19	1.29	3.79	3.19	4.29	
		2.19	4.09	2.99	2.79	3.35	4.09	3.39	4.39	
		1.99	3.79	3.03	1.19	1.29	3.99	3.39	4.39	
		2.19	4.09	3.00	2.99	4.24	4.29	3.59	4.49	
		1.99	3.69	3.03	1.19	1.29	4.09	3.39	4.49	
		2.19	4.01	3.02	3.79	2.19	3.03	3.49	4.39	
		1.99	3.69	3.04	1.19	1.29	4.10	3.40	4.49	
		?	?	3.03	?	?	3.09	?	?	
		1.89	3.69	3.12	1.19	1.39	4.49	3.39	3.89	
							4.50	3.90	4.61	
		1.85	3.54							Turned TM off
							Comm.	will not	step	Turned TM on
		Only channel reading			4.49			4.49		Turned lights on
								4.44		
		1.90	3.79	3.19	1.20	1.36	3.69	3.99	3.49	#2=10°C;#6=10°C
		1.92	3.80	3.19	1.21	1.39	3.80	3.94	3.83	
		1.92	3.80	3.16	1.23	1.39	3.78	3.93	3.82	
		1.92	3.81	3.13	1.23	1.39	3.75	3.91	3.60	
		1.93	3.82	3.10	1.21	1.39	3.69	3.79	3.59	
		1.94	3.86	3.12	1.23	1.38	3.66	3.76	3.51	Occasional
		1.94	3.85	3.13	1.24	1.38	3.64	3.75	3.51	trouble with
		1.94	3.86	3.12	1.23	1.38	3.62	3.75	3.49	TM stepping

B